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A FEASIBILITY STUDY OF A COMMUNICATIONS SATELLITE FOR DEEP-SPACE MONITORING

by George W. Clemens, Jr., Alfred C. Mascy, and Duane W. Dugan

NASA Headquarters, OART
Mission Analysis Division
Moffett Field, Calif.

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SUMMARY

The advantages and feasibility of using a single Earth-orbiting satellite to serve primarily as a communications relay between Earth and space vehicles on interplanetary missions are examined. Possible advantages over ground-based antennas for this purpose are shown to include greater information rate and/or reductions in size and weight of communications equipment and of power supplies on board interplanetary spacecraft; continuous tracking of and communication with deep space vehicles by means of only one orbiting station rather than by multiple ground-based antennas; and immunity to weather, atmospheric noise, and ionospheric disturbances. In addition, such an antenna could serve as a navigation beacon for returning manned vehicles, as a system for calibrating Earth-based radio facilities, and as a facility for radio- and radar-astronomical observations, for the search of galactic intelligence, and for radio frequency studies.

A sample application of the concept to a typical manned mission to Mars is made in order to assess its feasibility. Included in the investigation of problems inherent in the use of an Earth-orbiting satellite as a communications relay station are orbit selection, attitude control, antenna reflector structures, equipment-module structure, power supply, micrometeoroid and radiation shielding, propulsion systems, logistics support, and launch-vehicle requirements.

Results of the study indicate that none of the problems examined in the sample application appears to preclude the realization of advantages cited for an orbiting communications relay station. Areas in which further study and research would be particularly advantageous to the implementation of the subject concept are defined in the study.

INTRODUCTION

Missions, manned or unmanned, to other planets of the solar system present severe communication problems. The distances over which transmission of information must be made are very large, so that on-board power requirements tend to be high and communications equipment large and heavy. In addition, mission durations are on the order of years, so that maintenance of

continuous tracking of and communication with interplanetary vehicles becomes of concern, particularly so in the event of concurrent missions.

Current solutions to these problems involve the construction of a number of large ground-based antennas spaced around the Earth at intervals of approximately 120° in longitude. In view of the expected increase in demands on deep-space tracking and communications as exploration of the solar system is broadened, it is believed to be worthwhile to consider alternative or supplemental methods of meeting such demands.

The purpose of this study is to investigate the feasibility of providing continuous communication with interplanetary spacecraft by the use of an Earth-orbiting antenna. The possibility of employing transmission frequencies much higher when outside the Earth's atmosphere than when within it offers a number of advantages not available to ground-based systems. These advantages include increases in information rate and/or savings in vehicle weight. Likewise, the possibility of utilizing the natural precession associated with satellite orbits about an oblate planet to prevent occultation while tracking interplanetary vehicles further invites examination of the concept.

The scope of the study included manned applications with corresponding logistic support for maintenance, replenishment, and reliability. However, it was found that the increased weight (chiefly of shielding) to protect manned operations required the use of one or more Saturn V launch vehicles to accomplish the mission. Since manned operations do not enhance the basic communication capability of the system, the results of the related studies are not included in the text although the figures and tables retain some of this information for reference.

In the following sections, advantages of employing the high transmission frequencies permissible outside the terrestrial atmosphere are examined in terms of information rate and power requirements of a mission vehicle. Next are considered the problems associated with maintaining continuous communication with vehicles on interplanetary missions by means of an Earth-orbiting antenna. These problems include the selection of a satellite orbit which provides, in a nearly optimal manner, the precession rates, inclinations, and altitudes required in a given application with radiation hazards taken into account. Other sections deal with attitude control; propulsion systems; antenna configurations and their construction; equipment module configurations and construction; power supply; micrometeoroid and radiation shielding; and launch-vehicle requirements.

Two possible types of orbiting communications station are considered. These are: (a) a receiver-relay system in which data from interplanetary spacecraft are received, stored, processed, and relayed to Earth stations on command (command signals to the vehicles and to the station would be transmitted by selected ground sites); (b) an unmanned transmitter-receiver-relay station in which a transmitter function is included to relay commands and information from Earth to spacecraft.

POSSIBLE ADVANTAGES OF USING AN ORBITING ANTENNA

One severe constraint on the selection of radio frequencies for continuous communications between ground-based facilities and deep-space vehicles is imposed by the terrestrial atmosphere. Above the atmosphere, a much wider range of frequencies is available for application to space-to-space communications. A simple example will serve to point out some possible advantages inherent in the use of frequencies higher than those currently used in the DSIF.

Consider two communications links both between a deep-space vehicle and Earth-orbiting facilities which are identical except for the frequencies employed. One form of the basic equation governing a communications link is

$$P_R = \frac{P_T K_T A_T K_R A_R f^2}{(4\pi R)^2 c^2} \qquad (1)$$

where

P_T, P_R transmitted and received power

K_T, K_R transmitting and receiving antenna design constants

A_T, A_R actual areas of the transmitting and receiving antennas

R range or distance between transmitting and receiving antennas

f frequency

c speed of light

With the assumption that one link operates at 2295 MHz (frequency adopted for DSIF) and the second at 10,000 MHz, a number of comparisons between the two links can be made with the aid of equation (1). For example, with a given transmitted power from the spacecraft antenna, the power received at the Earth-based receiver employing the 10,000 MHz frequency is nearly 19 times that received at the antenna operating at 2295 MHz. From other points of view, if the power received is fixed at some level, and if the higher frequency system is used in place of the lower frequency system, then the transmitted power required could be reduced 19-fold, the information rate increased 19 times, the limiting range increased by a factor of 4.36, or the spacecraft antenna diameter reduced 4.36 times. In practice, a combination of the foregoing advantages would likely be made. In any case, the use of practical frequencies in the X- (5.2-10.9 GHz), K- (10.9-36 GHz), and Q (36-46 GHz) communication bands above the atmosphere can contribute significantly to reducing interplanetary vehicle weights and increasing information rates over those now possible with frequencies currently used by the ground-based facilities. Associated with the use of higher frequencies is the reduction in beamwidth. This narrowed beam increases the requirement for pointing accuracy. A detailed study is required to assess the over-all effects of narrow beamwidth

but is not included in this discussion. Additional benefits may also be realized from an increase in efficiency of operation at the higher frequencies, and from reduced weight and volume of high-frequency components. Other advantages which result from operating a communications station above the terrestrial atmosphere are discussed in reference 1. Chief among these is the freedom from much of the noise generated within the atmosphere and by the warm Earth. Absence of atmospheric weather also eases structural and maintenance problems.

Although intended primarily as a link in communications between Earth and vehicles on interplanetary missions, an orbiting communications facility properly modified could serve possibly other useful purposes when not otherwise employed. Among the observations and applications which might benefit from the capability of a properly oriented orbiting antenna for nearly continuous surveillance are the following: radio and radar astronomy; radio-frequency investigations; search for galactic intelligence, where continuous observation is invaluable (see ref. 2); calibration of ground-based radio facilities; and a navigational beacon to aid deep-space spacecraft returning to Earth.

Apart from the advantages and benefits already discussed, there is one particular feature which alone might make the concept of an orbiting deep-space communications station worthy of consideration. This is the potential capability of a single orbiting station to perform the function of continuous monitoring of interplanetary flights which presently requires multiple (three or more) ground-based stations of the DSIF network. Concurrent space missions could more easily be carried out if both the DSIF and an independent orbiting communications station were available for required tracking and recording of data from multiple spacecraft.

FEASIBILITY ANALYSIS

In this portion of the study, a number of factors related to the use and operation of an Earth-orbiting radio station for deep-space communications are investigated.[1] The purpose here is twofold: first, it is essential to know whether there are any insurmountable problems which would preclude the implementation of the concept; second, it is believed useful to define problem areas in which further work and research are needed.

The first aspect of the subject concept to be discussed concerns the selection of orbits for the relay station. The orbital characteristics of inclination and altitude are important since they can affect the amounts of weights required for radiation shielding of sensitive electronic components and/or for propulsion if used to modify the orbit during the application of the communications station to a given deep-space mission.

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[1]The following acknowledgements are made for preparation of certain sections of this study: Mr. John A. Wyss, antenna structures; Mr. Joseph L. Anderson, equipment module; Mr. J. Michael Coogan, radiation shielding; and Mr. Robert E. Slye, launch-vehicle capabilities.

The orbital characteristics found feasible for the example mission are then used as a basis for estimating requirements for attitude control. Antenna structural problems are examined next. In turn, attention is given to the equipment module, the power supply, micrometeoroid shielding, radiation shielding, orbital propulsion systems, and finally, to launch-vehicle requirements. The discussions of these problems are less detailed than that of orbital selection.

For the purposes of this part of the study, the following characteristics are postulated for the orbiting communications station. A paraboloidal antenna having a diameter of 26 m is assumed for both transmission and reception of information to and from an interplanetary vehicle. An operating frequency of 10,000 MHz is selected as possibly a reasonable compromise among such communications factors as bandwidth, beamwidth, and accuracy tolerances of the antenna reflector. The effective area is conservatively assumed to be 0.5 of the actual antenna area. For use as a receiver-relay device only, the power requirements for communications are estimated as 75 to 100 W for continuous reception of information from a deep-space vehicle, with a peak value of approximately 1 kW during periodic periods of transmission of data to ground-based stations. For transmission from relay station to Earth, four equally spaced paraboloidal antennas 1.2 m are considered adequate. If transmission is contemplated at information rates up to and including those required for adequate voice communication to manned interplanetary spacecraft at distances up to 10 A.U. (1.5×10^9 km), 30 kW of transmitted power might be required. This figure assumes that a deep-space vehicle is equipped with a paraboloidal antenna of 5.5 m in diameter. Two-way voice communication over distances greater than about 10 A.U. probably represents a requirement beyond that contemplated for manned missions within the next few decades; hence, the figure of 30 kW for transmitted power is regarded as adequate for all foreseeable applications of an orbiting transmitter-receiver-relay station. Further analysis of the many trade-offs possible among communications parameters could no doubt result in a more efficient communications system than the one hypothesized here.

Orbit Selection

As noted previously, it is essential to its stated purpose that an Earth-orbiting antenna be capable of maintaining line-of-sight communication with an interplanetary vehicle with little or no interruption. In addition, the necessity for constant reacquisition of the communication link should be minimized. The selection of an orbit which enables an antenna-bearing satellite to fulfill this requirement is necessarily a compromise among a number of conflicting constraints. For example, inasmuch as nearly all interplanetary trajectories lie essentially in the plane of the ecliptic, the plane of the antenna orbit should preferably be nearly normal to the ecliptic to minimize the possibility of an occultation of the antenna by the solid Earth or its atmosphere as viewed from a deep-space vehicle. However, the contemplated use of precession of the antenna orbit to match more or less closely the moving line of sight to the space vehicle requires that the antenna orbit be inclined to the terrestrial equator at angles i dictated by the precession rates required in a given application, and by the altitude of the orbit. Because

the equatorial plane is inclined nearly 23.5° to the ecliptic, the inclination of the antenna orbit with respect to the ecliptic could vary by nearly 47° (between $90^\circ - i + 23.5^\circ$ and $90^\circ - i - 23.5^\circ$) during an interplanetary round-trip mission. In addition, the line of sight to the target vehicle is likely to lead or lag the projection of the normal to the antenna orbit upon the ecliptic during the mission. Hence, occultation of the antenna could occur in some instances. Increasing the orbital altitude appears to be one obvious means of avoiding occultation; however, the precession rate decreases with altitude, so that for a given desired rate the (acute) angle of inclination of the orbit with respect to the equator (and, hence, to the ecliptic) must be decreased to compensate for the effect of the increase in altitude. Thus, the effect of increasing altitude in an effort to avoid occultation is offset to some extent. Furthermore, the choice of orbital altitude is subject to constraints. The altitude should be high enough to avoid aerodynamic drag effects, yet not so high as to penetrate so deeply into the Van Allen radiation belts that weight of radiation shielding becomes prohibitive. Orbits beyond the radiation belts do not appear attractive, since the launch requirements, not only initially but also for subsequent logistic support, are large compared with those of orbits below the belts.

The geometry associated with the antenna orbit is shown in figure 1. The orbit is shown inclined to the equatorial plane at an angle i; the ascending node is at right ascension Ω (measured from the vernal equinox Υ in the equatorial plane) or at heliocentric longitude l (measured in the ecliptic plane). The angle to the instantaneous line of sight between the center of the Earth and the target vehicle is δ and in practical applications can be considered to be measured in the plane of the ecliptic. Because the ratio between the distance from Earth to an interplanetary spacecraft and the radius of a near-Earth satellite is, for all but short periods following launch or preceding return to Earth of a spacecraft, several orders of magnitude greater than unity, the reference line of sight can, with negligible error, refer to the line of sight between the orbiting antenna and the target vehicle. As remarked earlier, the line of sight to the space vehicle may lag or lead the line P perpendicular to the antenna orbit by an azimuth angle φ (measured in the ecliptic plane); likewise, the line of sight may be above or below the normal to the orbit by an elevation angle η (measured in the plane normal to the ecliptic and containing the line P). As shown in the figure, the resultant of these last two angles constitutes the total look angle θ.

To assess the effect of the antenna orbit on weight penalties, an application of the concept is made to a typical manned landing-and-return mission to Mars during the 1980 opposition period. The outbound and return trajectories of one such mission are illustrated in figure 2 (see ref. 3). Briefly, the profile of this mission is (1) launch from a near-Earth orbit into a 251.5-day flight to the vicinity of Mars; (2) propulsion braking into a parking orbit about the planet; (3) launch from the Martian parking orbit after a 27-day stay into the return leg of 208.5 days. In figure 2, the line of sight from Earth to spacecraft is shown for a number of elapsed times since launch. In figure 3, the variation of the related angle δ during this mission is given. The dotted line shown with a slope of 0.5°/day is drawn to indicate how a single precession rate might be used to approximate the variation of the line-of-sight angle during the mission. Similar data for another

round-trip mission were calculated. In this latter mission, atmospheric braking is used to establish a parking orbit about Mars, a shorter staytime of 7 days is assumed, and the optimum mission time is 427 days, or two months less than the first mission. Here a lower single precession rate of 0.4°/day gives an approximation to the history of the angle δ. At any given time after launch of the spacecraft from Earth orbit, the difference between the solid curve for δ and the dotted line in each case is very nearly equal to the angle φ if the antenna orbit is oriented initially so that $l_i = \delta_o \pm 90^{\circ}$, where δ_o is the intercept of the dotted line on the δ axis. The equality is approximate since the rate of precession $\dot{\Omega}$ and the angle Ω are in the equatorial plane rather than in the plane of the ecliptic in which δ, l, and φ are measured. The discrepancies, although never large, are taken into account in subsequent calculations.

The typical mission of figure 3 is selected for an application of the concept of an orbiting antenna since its more rapid average rate of change of line of sight appears to present a somewhat more difficult problem. Orbit selection involves searching for combinations of orbital inclination and orbital altitude which will result in precession rates adequate for tracking the space vehicle with little or no occultation and at the same time minimize weight penalties. Figure 4 shows the effects of orbital inclination and orbital altitude upon the rate of precession for circular orbits. As an example, if the precession rate is taken as 0.5° for the duration of the mission, and the orbital inclination is assumed to be 95°, the orbital altitude of a circular orbit must be very nearly 1080 km.

From considerations of the geometry involved, it can be shown that the minimum altitude h_{min} at which occultation can be avoided can be calculated from

$$h_{min} = \left(R_{\oplus} + h_{ion}\right)\left(\frac{1}{\cos \theta} - 1\right) + h_{ion}$$

$$\cos \theta = \cos \varphi \cos \eta$$

where

$R_{\oplus}$ radius of Earth

h_{ion} assumed altitude of disturbing ionosphere

φ angle by which the line of sight to vehicle leads or lags the perpendicular to the antenna orbit

η complement of angle of inclination of antenna orbit with respect to the ecliptic plane

θ total look angle

The angle η is found from

$$\tan \eta = \frac{1 + \cos l \tan I \tan i \cos b}{\tan i \cos b - \tan I \cos l}$$

$$\sin b = \frac{\sin I \sin l}{\sin i}$$

where I is the obliquity of the ecliptic. The angle l, the heliocentric longitude of the moving ascending node on the ecliptic, is calculated from the equatorial angles i and Ω as follows

$$l = \tan^{-1} A + \tan^{-1} B$$

$$A = \frac{\cos(1/2)(i+I)}{\cos(1/2)(i-I)} \tan \frac{1}{2} \Omega$$

$$B = \frac{\sin(1/2)(i+I)}{\sin(1/2)(i-I)} \tan \frac{1}{2} \Omega$$

and

$$\Omega = \Omega_i + \dot{\Omega} T$$

where Ω_i is the initial right ascension of the ascending node of the antenna orbit corresponding to the initial value of l ($l_i = \delta_o \pm 90^\circ$) in the ecliptic; $\dot{\Omega}$ is the rate of precession of the line of nodes in the equatorial plane; and T is the time after launch of the interplanetary spacecraft from Earth orbit.

The angle φ is calculated from

$$\varphi = l - \delta + 90^\circ \quad \text{for} \quad l_i = \delta_o - 90^\circ$$

$$\varphi = l - \delta - 90^\circ \quad \text{for} \quad l_i = \delta_o + 90^\circ$$

The resultant of φ and η (the total look angle θ) is shown in figure 5. This is the angle through which the antenna must be rotated in order to point at the target vehicle. The rate of change $\dot{\theta}$ of the look angle is given in the case of the example mission in figure 6. This rate is usually less than 1°/day, and for long periods it is only 0.04°/day. The maximum rotation rate required of the orbiting antenna is more than two orders of magnitude lower than that usual for Earth-based antennas, and thus inertial loads would be correspondingly lower. It should be noted that the pertinent angular rates are those about axes fixed in the orbiting station, since these are the rates which must be provided by the attitude control system. However, the rates about any antenna axis would not exceed the rate of change of the total look angle.

The minimum altitude for no occultation is presented in figure 7. Occultation is considered to occur if the line of sight to the target passes through the ionosphere. For the purpose of this study, the ionosphere boundary h_{ion} is taken as 80 km above the Earth. For the antenna orbital altitude of 1080 km and single precession rate of 0.5°/day ($i = 95^\circ$), figure 7(a) shows that occultation occurs during three periods of about 30 days each in

the example mission (i.e., the value of h_{min} exceeds the orbital altitude h = 1080 km). During periods of occultation, communications between the relay station and space vehicle would be impaired or prevented for generally only a few minutes out of each orbital period of about 100 minutes. Nevertheless, interruptions in communications may be regarded as serious near the end of the mission since the radio station could be advantageously used as a navigation beacon to guide the returning vehicle to an atmosphere-entry corridor. Increasing the altitude to 1300 km while maintaining the same rate of precession is shown in figure 7(b) to be effective in eliminating occultation except for the first two days of the mission. Occultation so near the Earth on the outbound leg of the mission is not considered to be serious. More important is a possible weight penalty incurred by increasing the orbital altitude. As discussed in a later section, the structure of the relay station provides sufficient shielding (about 1 g/cm^2) to prevent significant radiation damage to electronic components in the time period of the example mission (487 days) for altitudes well beyond 1300 km. Hence, for unmanned orbiting radio stations no weight penalty for shielding results from increasing the orbital altitude from 1080 to 1300 km. Launch requirements, of course, are somewhat greater for the higher orbital altitude. These are discussed in a later section. On the other hand, shielding requirements for a manned station increase sharply from about 17 g/cm^2 at 1080 km to nearly 60 g/cm^2 at 1300 km, based upon a period of 90 day's exposure.

An alternative method of avoiding occultation without increasing the altitude above 1080 km is to divide the mission period into two or more parts, with each part characterized by an individual rate of precession. If an initial orbital altitude is maintained, a change in orbital inclination is required for each change in precession rate. Figure 7(c) illustrates the results of making a plane change in the antenna orbit 140 days after the start of the mission from an initial 93° ($\dot{\Omega}$ = 0.3°/day) to 95.5° ($\dot{\Omega}$ = 0.55°/day) while maintaining a constant orbital altitude of 1080 km. Again, as in figure 7(b), occultation is avoided except for the first two or three days, when an occasional short interruption in communications is not regarded as serious. The plane change of 2.5° involves a velocity increment of about 0.32 km/sec.

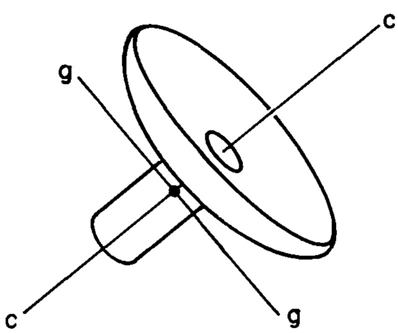
Another method for avoiding occultation at the lower altitude by applying propulsion to rotate the line of nodes of the orbit was also investigated; however, such maneuvers proved to be more costly, in terms of velocity, than the orbital plane change.

The foregoing examples of solutions to problems involved in selecting an antenna orbit for a particular interplanetary mission are in no way to be regarded as optimal solutions. Other combinations of orbital parameters, together with initial orientation of the line of nodes of the orbit can likely reduce the magnitude of required plane changes and, hence, the amount of weight penalty. It is concluded from the results obtained so far that communications stations in near-Earth, near-polar orbits can maintain essentially continuous line-of-sight communications with interplanetary spacecraft having average rates of change of the line of sight at least as large as 0.5°/day.

Attitude Control

The primary objective of attitude control of the communications satellite is to keep an antenna accurately and continuously pointed at a target vehicle. To accomplish this task, the antenna must be oriented in accordance with changes in the required look angle, θ. In addition, the attitude control system is required to compensate for various disturbing torques.

As noted earlier, the rate of change of the look angle would be small, being as large as about 1°/day only at the beginning and at the end of the example mission. Accordingly, the accelerations would also be small, about $0.05^{\circ}/\text{day}^2$ near the beginning and end, and far less during the middle portion of the mission. To estimate the maximum torque which might be required to supply this acceleration, the mass moments of inertia of the transmitter-relay station are used. The approximate moments of inertia calculated are I_c = 34,000 kg-m-sec^2 about the central axis c to c (sketch (a)) and I_g = 38,300 kg-m-sec^2 about the tumbling axis g to g. The required torque associated with changing the look angle is about 0.45×10^{-8} kg-m. As will be seen, these torques are much less than those required to maintain the angle accurately.



Sketch (a)

At the altitudes of interest (1000-1300 km), the predominant disturbing torques acting on the station would be those due to the Earth's gravity gradient and to internal movements of instruments, control mechanisms, etc. Because of the asymmetry of the example station configuration ($I_g - I_c$ = 4360 kg-m-sec^2) the Earth's gravity gradient would produce approximately 0.0032 kg-m of torque. This value represents an average gravity-gradient torque, since the magnitude would change during the mission as the latitude and longitude components of the look angle varied. An additional 0.0032 kg-m of torque is assumed to take into account moments due to internal movements, atmospheric drag, solar radiation pressure, Earth's oblateness, micrometeoroid impacts, etc. Whereas the net gravity-gradient torque is taken as the root-mean-square value about two axes, the net torque due to random disturbances is taken as the rms value about three axes. Estimates indicate that the moments are very nearly the same for orbital altitudes of 1080 and 1300 km. With two thrustors located 7.6 m apart along each axis, the required thrust would be 0.403 g.

Since the torque due to gravity gradient appears to be significant, the possibility of using this effect for vehicle stabilization was considered. This technique was not found to be attractive, however, primarily because an antenna must look below the horizon on one side of the orbit and above the horizon on the other. With a vehicle stabilized by gravity gradient the antenna attitude would have to be cycled through plus and minus the look angle in each orbit revolution. A more desirable approach may well be to design a vehicle with moments of inertia that are nearly equal about all axes in order to reduce the gravity-gradient torque. Additional study in this area is required.

For the example mission, weights of propellants were calculated for each of three requirements in attitude control, namely, changes in the look angle, counteraction of gravity-gradient moments, and stabilization against random disturbances. Two different types of thrustors were postulated, one a moderately high-thrust chemical system with a specific impulse of 250 sec, the other a continuous low-thrust device (e.g., an ion engine) having a specific impulse of 5000 sec. The latter type is considered to be feasible in case a nuclear-reactor power supply is available to supply the large power requirements of a transmitter-relay radio station in orbit. The results are shown in the table below. The advantages of using ion thrustors of high specific impulse for attitude control are apparent from this table.

PROPELLANT WEIGHTS FOR ATTITUDE CONTROL
[Percentage of Gross Weight in Orbit]

Requirement \ Propulsion system	Medium-thrust chemical	Low-thrust electrical
Change look angle	Negligible	Negligible
Resist gravity gradient	1.7	0.08
Random disturbances	2.0	.10

Should neither the chemical nor ion engine be available or suitable for the purpose of attitude control, currently used devices such as cold-gas reaction jets could be employed. However, the weight of gas and associated tankage would be many times larger in this case than if even chemical engines were used.

Propulsion Systems for Orbital Use

As noted earlier, one method of avoiding occultation of the target vehicle involves an orbital plane change. Various missions would no doubt require different altitudes and inclinations and orientation of the orbit to be established for the orbiting station. Hence, some propulsive device and propellant supplies would be required for orbital maneuvering.

Two classes of propulsion systems are considered here in order to obtain estimates of weights required for propellants in the case of the plane change of 2.5°. One is a chemical rocket engine having a specific impulse of 250 sec; the other is an electric ion engine with a specific impulse of 5000 sec. The latter propulsion system, like the ion thrustors suggested for attitude control, is considered to be practical only in the event that a nuclear reactor is available to supply the power requirements for a transmitter-relay communications station. If the chemical propulsion system is used to accomplish

the plane change at the altitude of 1080 km, the propellant weight is 12.2 percent of the gross weight in orbit; if ion propulsion is possible, the propellant requirement and engine would be only 1 percent. These weights are included in the total propellant weights (including those for attitude control) given in table I for various types of orbiting communications stations considered here.

Antenna-Reflector Structures

One of the most important criteria in the design of reflectors is achievement of accurate paraboloidal surfaces in order to achieve the desired gain. For an orbiting antenna, these accurate surfaces must be obtained with structures that can be packaged compactly for launch. Three types of collapsible antennas are considered in the present study: (1) a fold-out, all-metal type; (2) an antenna having a combined metal core and inflated outer panel; and (3) an inflated and rigidized structure.

For antenna dishes of any size, good focusing and efficiency are obtained when the surface irregularities do not exceed about one-twelfth of the operating wavelength, λ (ref. 4). For example, at a frequency of 3,000 MHz, the required accuracy is ± 0.83 cm; at 10,000 MHz it is ± 0.25 cm. With current structural techniques, the ratio of the rms surface deviation to the diameter is about 10^{-4} (ref. 4). On this basis, the surface inaccuracies would be about ± 0.25 cm for an antenna 26 m in diameter. It follows that such an antenna could be operated at frequencies up to about 10,000 MHz by using the $\lambda/12$ relationship cited.

Fold-out antennas.- Figure 8 shows a possible design for an all-metal, fold-out antenna having a diameter of 26 m. The antenna is conceived to consist of a fixed center section 6.1 m in diameter supporting both a tower for a Cassegrainian reflector and a short tower for a horn mount. Thirty panels are designed to be folded rearward within the confines of a cylinder having a diameter (6.5 m) compatible with a Saturn-class booster. A central core 2.44 m in diameter is intended for use as an equipment module.

To evaluate panel deflection a simple beam analysis is used. The load condition is assumed to be due to surface gravity so that the antenna would have the strength and rigidity necessary for ground testing with minimum support. This assumption is conservative, since a lighter antenna could be used in space and could be ground tested with an appropriate support system. The structural material is considered to be large-celled aluminum honeycomb. Calculations indicate that under these conditions, the upper and lower skins should be about 0.08 cm thick. Analysis also suggests that the skins should be tapered in thickness, and that tapered supporting ribs could be used. The resulting thicknesses of skin correspond to a unit weight of about 4.9 kg/m^2. With the ribs and tower structure included, the total weight for the paraboloid is estimated as 3900 kg. For this type of construction, the weight W_a (kg) of corresponding antenna structures of any diameter D (meters) can be approximated by

$$W_a = 30.6\,D + 4.65\,D^2$$

The first term primarily represents the weights of the tower and honeycomb, while the second term accounts for skin weight. The variation of weight with antenna diameter is presented in figure 9. Although the weights given by these equations are only approximate, they are considered adequate for comparison with those estimated for other methods of construction.

Antenna with combined metal core and inflated outer panel.- An inflatable antenna structure has a packaging advantage, particularly when antenna diameters as large as 61 m are considered. However, some experience in industry has indicated that at present it is not possible to maintain the desired surface accuracy for diameters over 12 m. With a central rigid core retained, a 26-m diameter can be achieved with an inflatable outer section having a radial dimension of about 10 m. A double surface structure with ribs 1.5 m deep at the root is chosen as a design concept. Assessment of the feasibility of achieving sufficient surface accuracy with such a large inflatable structure requires a detailed membrane analysis which is beyond the scope of the present study. Possible weight savings with this structural arrangement are indicated in figure 9 where weights for this approach and the fold-out approach are compared. For an antenna of 26 m in diameter, the all-metal rigid antenna has an estimated unit structural weight of 6.8 kg/m^2, whereas the partly inflatable structure has a unit weight of about 4.9 kg/m^2.

Inflated and rigidized structures.- Perhaps the greatest weight savings could be realized in the construction of large reflectors for use in orbit by extending the application of inflatable-type construction discussed in the preceding paragraph to the entire antenna reflector. In the case of antennas having diameters as large as 61 m, the use of inflatable and rigidized structures may be necessary, primarily because they must usually be packaged in relatively small volumes for launching. On the other hand, a number of difficulties can be anticipated with such structures. For example, the unfolding process and ground testing might well pose major problems. Perhaps the most serious problem is obtaining surface accuracies commensurate with the desired gain of antennas as large as 26 m or more in diameter. No satisfactory solution to this problem has yet been found.

Equipment Module

In the radio station conceived here, an equipment module forms the structural backbone of the entire vehicle. This module is considered to be cylindrical in shape, about 2.5 m in diameter and 10.7 m long. The antenna of the all-metal, fold-out type, is assumed to be in a furled and stowed arrangement along the outer cylindrical surface of the equipment module during launch and flight through the atmosphere and prior to deployment. One end of the module provides attachment to the booster; the other end serves as the antenna base and attachment point for the receiver-horn structure. Figure 10(a) shows how the study vehicle might be mounted on the booster. The launch arrangement for the vehicle is conceived to be similar for both the receiver-relay and the transmitter-receiver-relay configurations. A shroud protects the antenna from the aerodynamic noise and buffetting loads during launch and flight through the Earth's atmosphere. It is designed to separate

from the spacecraft early during second stage booster burning. The antenna is to be erected after the station attains its operational orbital altitude.

Figure 10(b) shows a vehicle in a conceptual operational arrangement as a receiver relay. Four Earth communications antennas are mounted at 90° positions on the base of the cylindrical portion, and the primary and secondary attitude-control and orbit plane-change motors are located at the ends of the equipment module. The structure of this module consists of a monocoque-type meteoroid shield and a honeycomb-reinforced and insulated inner pressure vessel. A structural weight of about 1 g/cm^2 results for the module. Temperature control within the module is believed to be possible by a passive system employing conduction through the module structure without the necessity of maintaining pressurization during normal operation. The module contemplated here has a volume of about 57 m^3 and an internal headroom diameter of 2 m.

Figure 10(c) shows an operational arrangement for a conceptual transmitter-receiver-relay vehicle. The equipment module and antenna are similar to those of the receiver-relay vehicle; however, as discussed in the following section, the greater power demand is best provided by a nuclear reactor, generator, and radiators. The reactor unit is shadow shielded to protect the equipment module and the active antenna components. The coolant-radiator conical structure supports the generator, reactor, and shielding. The radiators are designed to have sufficient area to fulfill the cooling requirements of both the reactor and the communications equipment. Supplemental attitude-control and orbit-change propellants are assumed to be stored within the radiator structure.

It is estimated that the basic equipment module for an unmanned receiver relay might weigh about 1130 kg. For an unmanned transmitter-receiver relay, an additional 320 kg would be required so the weight would be 1450 kg.

Weights of the module components and of possible equipment are listed in table I.

Power Supply

The power requirements would be dictated by the type of communication system employed and the source of power would in turn be governed by the application. Unmanned receiver-relay stations would require only about 75 to 100 W of power for reception and a nominal 1 kW peak power capacity for periodic transmission to Earth. An additional 150 W should be provided to satisfy spacecraft needs, such as battery recharging; gyroscopes; instrumentation; attitude-control mechanism, including star trackers; and any necessary pumping for thermal control. Accordingly, the power requirements for an unmanned receiver-relay station are estimated to be a total of 250 W for continuous operation, with separate provisions for 1000 W for short-duration transmission of stored data to Earth.

Among the power sources considered here for a relay station are solar cells and radioisotopes. Solar-cell panels present a large area vulnerable to micrometeoroids and also require constant orientation to the Sun. In addition

to the attitude-control problems, it would be necessary to establish an orbit which would be simultaneously and constantly in view of both the Sun and an interplanetary vehicle. Among the advantages of radioisotopes are compactness and the reliability demonstrated by the years of operation of SNAP-9A power sources aboard the Transit satellites. For these reasons, radioisotopes were considered as the primary source of power for the simple relay station.

The requirement for 250 W of continuous power could be met by multiple SNAP-9A units. Ten such units, having a total weight of about 114 kg, would be required. Power for relay transmission to Earth could be provided by a silver-cadmium ceramic-sealed secondary battery. To provide power for 1 hour of relay transmission per day (i.e., 1000 W-hr) during a 50-percent discharge, the battery would weigh about 34 kg. With a 50-percent discharge once a day in a temperature environment of 25° C, this battery has a life of 600 cycles.

On the other hand, power requirements for a transmitter-receiver-relay station would, of course, be much greater. In the present study, as noted earlier, a transmitted power of 30 kW from the orbiting antenna is assumed to be adequate. With a conservative efficiency of 20 percent, an electrical input power of 150 kW would be required. This magnitude of power can best be realized for space applications by the use of nuclear reactors.

A nuclear reactor with an output of approximately 0.8 thermal megawatt could provide the required heat for a turbo-generator system. A radiator area of about 28 m^2 is required to dissipate the waste heat. It is estimated that such a power source would weigh approximately 10 to 15 kg/kW with an additional 3 kg/kW for radiation shielding. The total weight is estimated to be 2500 kg. This figure includes the weight required for meteoroid shielding of the radiator.

As noted in an earlier section, this nuclear power supply might also be used to reduce the propellant expenditures associated with required orbital maneuvers.

Micrometeoroid Hazard

The penetration theory developed by Summers and Charters (ref. 5) was used together with Whipple's "1963 A" estimate of the particle flux (ref. 6) to determine the possible micrometeoroid hazard to an orbiting antenna. The structural material of the antenna was taken as aluminum 0.081 cm in thickness. If the diameter is taken as 26 m, the total exposed area of the antenna dish, front and back, is 1121 m^2. For the typical mission time of 487 days, approximately 1750 penetrations might occur. This number corresponds to 3.6 penetrations per day. It may also be shown that the diameters of these punctures are less than 0.03 cm.

It therefore appears that micrometeoroids present negligible structural and operational problems to the antenna reflector itself. Likewise, the present analysis indicates that the structure of the equipment module described in an earlier section provides adequate protection of electronic components

and other equipments against damage from micrometeoroids. On the other hand, all radiators required for thermal control must be protected.

Radiation Shielding

In an unpublished paper, Adams and Holly of the Air Force Weapons Laboratory utilize the computer codes of Barton, Keister, and Mar (ref. 7) to predict radiation doses that would be received behind various shielding configurations while orbiting the Earth at different altitudes and inclinations. This work took into consideration the effects of trapped protons and electrons (including bremsstrahlung) and solar protons.

Adams has extended this effort for a range of altitudes of interest to this study in the case of simple spherical shields of 10 and 20 g/cm^2 of aluminum. The results in terms of rad/day received as a result of the trapped radiation are shown in the first table below. The second table shows the results in rad/duration of flare for various solar-proton events.

TRAPPED RADIATION DOSE*
FOR ORBIT INCLINATION = 90°

Shielding / Orbit altitude	Primary proton dose		Primary electron dose	Bremsstrahlung dose		Total dose	
	10 g/cm^2	20 g/cm^2	Both cases	10 g/cm^3	20 g/cm^2	10 g/cm^2	20 g/cm^2
900 km	1.6	0.83	0	0.54	0.44	2.1	1.3
1080 km	3.1	1.5	0	.93	.74	4.0	2.3
1300 km	6.2	3.2	0	3.8	3.1	10.0	6.3
1500 km	11.2	5.7	0	8.0	6.7	19.2	12.4

*Dose units are rad/day

From the data in the first table, the number of days permitted in circular 90° orbits at various altitudes can be calculated for different aluminum shield thicknesses and different total dose limitations. These relationships are illustrated in figure 11. (Dashed curves indicate extrapolations to greater values of shielding.) The dose limitations shown were chosen because they represent biological or equipment exposure limits. Recently defined NASA average yearly dose limits for humans include 27 rads for eyes and 233 rads at the surface of the outer skin of the entire body (ref. 8). Lehr, Tronolone, and Horton (ref. 9), in establishing functional damage in terms of absorbed dose for various materials and components, found that transistors fail under doses of 2×10^3 to 2×10^4 rads.

A nonsensitive type transistor would require no more shielding than is furnished by the structure of the equipment module (1 g/cm^2) to survive for 500 days at altitudes up to 1500 km.

From the solar-flare data given in the second table it is deduced that the total dosage received from even several flares with a specified shielding (e.g., 10 g/cm^2) at altitudes of interest here (1000 to 1300 km) during any 90-day period is likely to be about 10 percent or less than the dosage

received from the high-energy trapped particles of the radiation belts (i.e., about 50 to 60 rads compared with nearly 550 rads).

SOLAR-FLARE PROTON DOSE;* INCLINATION = 90°

Flare \ Shielding (Orbit)	Altitude = 1500 km		Altitude = 400 km	
	10 g/cm^2	20 g/cm^2	10 g/cm^2	20 g/cm^2
23 Feb. 1956 Duration = 7.77×10^5 sec	18.8	9.2	18.4	9.0
14 July 1959 Duration = 5.11×10^5 sec	7.8	1.7	7.4	1.6
12 Nov. 1960 Duration = 4.89×10^5 sec	15.6	5.48	15.2	5.3

*Dose units are rad/duration of flare.

Launch-Vehicle Requirements

The launch-vehicle requirements for an orbiting relay station depend upon the total weight of the vehicle, upon the characteristics of its orbit, and upon the location of the launch site. As discussed in earlier sections, the weights are affected by numerous considerations. These weights, summarized in the following table, were obtained by summing the appropriate component weights listed in table I. Unmanned stations might weigh approximately 5,800 to 6,700 kg if intended for relay functions only, and from 9,300 to 11,000 kg if facility for transmission is added.

EFFECTIVE ORBITAL WEIGHTS, kg
[26-m-Diameter Antenna]

Altitude	Method of attitude control and orbital changes	Weight conditon	Receiver relay unmanned	Transmitter relay	
				Unmanned	Manned (shielded)
1080 km orbit 2.5° plane change	Chemical propulsion	Dry Propellant Total	5600 1060 6660	9250 1750 11,000	37,500 7090 44,590
	Electrical propulsion	Dry Propellant Total		9250 110 9360	37,500 450 37,950
1300 km orbit No plane change	Chemical propulsion	Dry Propellant Total	5600 215 5815	9250 355 9615	87,700 3370 91,070
	Electrical propulsion	Dry Propellant Total		9250 17 9267	87,700 158 87,858

From the range of weights estimated for the various types of radio stations considered, the launch-vehicle capabilities required to place these weights into the desired orbits lie between those of such launch vehicles as Titan IIIC and Saturn IB. Because of range safety considerations, the usable launch azimuths from launch facilities at Cape Kennedy cannot exceed 160°. For the example mission to Mars, initial orbital inclinations of 93° to 95° were cited; the corresponding launch azimuths are nearly 184° and 186°. Hence, dog-leg trajectories are required in launching the communications station from Cape Kennedy. The reduction in payload capability with increasing orbital inclination, with an increase in altitude, and with azimuth restrictions is illustrated in figure 12 in the case of the Saturn IB launch vehicle. For launches from the facilities at the westcoast launch site, no dog-leg maneuver would be required. However, at present, the Saturn launch vehicles cannot be accommodated at the westcoast site. Use of the Titan IIIC vehicle might require a reduction in the diameter of the launch configuration of the communications station from 6.5 m considered necessary in the case of the 26-m antenna, to nearly 3 m.

A summary of launch-vehicle requirements for placing several types of communications satellites into one or another of two prescribed orbits is given in figure 13. The corresponding payload capabilities of three different launch vehicles are indicated in the figure. In the case of the Titan IIIC, launch from the Pacific Missile Range was assumed, with no restrictions on the launch azimuth; for the other two vehicles, a dog-leg maneuver entailing a yawing of the second stage was assumed to take into account a restriction to 160° in launch azimuth required at the Atlantic Missile Range.

DISCUSSION OF RESULTS

In the foregoing sections, some of the major problems associated with the implementation of the concept of an orbiting deep-space communications station were examined. To provide a realistic basis for assessing the feasibility of the concept, a hypothetical application of the station was made to a typical manned exploration mission to Mars.

The selection of an orbit for the communications satellite was guided by the average rate of change of the line of sight to the spacecraft during the major portion of the mission (0.5°/day). With a rate of precession of the orbital line of nodes of the same value, a combination of an altitude of 1300 km and an orbital inclination of 95.4° was found to avoid occultation by a small margin. In the case of the unmanned transmitter-relay station utilizing ion engines, the weight penalty for making the plane change of 2.5° was more than offset by the increase in payload capacity of the launch vehicle for the orbital altitude of 1080 km rather than for 1300 km. Further trials possibly could produce orbits somewhat more advantageous from the weight standpoint.

The relative magnitudes of component weights listed in table I for both manned and unmanned communications satellites tend to indicate study areas in which more effort would be most useful.

One major item of weight, particularly in the unmanned systems, is the large antenna of the all-metal foldable type. Other possible methods of construction were examined and appeared to offer lower weight and smaller packaged size; however, the achievement of sufficient accuracy of construction to assure the required gain of large antennas remains to be demonstrated with such methods.

For the transmitter-relay station, the weight of the large power supply constitutes a sizable fraction of the total weight. To furnish approximately 150 kW, a nuclear reactor would presumably be required with a resultant need for shadow shields. Provision should be made for shutting down the reactor during maintenance visits by technicians and for subsequent restart.

The combined weights of propellants for attitude control and for the orbital plane change are also quite large if chemical rocket engines are used. A capability for thousands of restarts would have to be developed for small chemical rockets intended for attitude control for long periods of time. The use of other methods such as cold-gas jets would increase the weight required for propellants and tankage many times over that required with the chemical rockets. On the other hand, in the case of the transmitter-relay station for which a large power supply is essential, the use of ion engines of high specific impulse could reduce the weight of propellants needed for attitude control by a factor of 20 or more in comparison with the chemical system. Similar savings in propellants used for orbital maneuvers could be effected, as shown in table I.

Several other weights, such as that of the equipment module and the aerodynamic shroud, are fairly well determined by the size of the antenna and volume requirements of equipment, and in some cases by the launch vehicle to be used.

The total orbital weights of the simple relay and the transmitter relay are shown in figure 13(a). Orbital altitudes of 1080 and 1300 km are assumed, and all stations are considered to be equipped with main antennas 26 m in diameter.

For comparison, the corresponding payload capabilities of three launch vehicles are indicated in the figure. Although it is noted that the simple relay vehicle is within the launch capability of the Titan IIIC at either altitude, the diameter of the launch vehicle is too small to accommodate the assumed all-metal foldable antenna. A smaller antenna would be required to be compatible with the launch vehicle, and/or a different type of construction which would permit smaller packaging for launch would have to be used. The unmanned transmitter-relay station in the lower of the two orbits might possibly be placed into orbit by a Saturn IB vehicle if ion engines could be used for the plane change of 2.5° and for attitude control.

Figure 13(a) indicates that the orbital weights of the unmanned transmitter-relay communications satellite are only slightly greater than the payload capabilities of the Saturn IB and of the Titan IIIC; inasmuch as there are distinct advantages in having a capability for transmission in the subject communications station, efforts to reduce the weights by one means or another

appear worthwhile. In making estimates of weights throughout this study, a conservative approach was adopted. Further consideration of some of the problems involved in designing various components might lead to more confident and somewhat lower estimates. In any case, one obvious way to reduce the orbital weight of the orbiting transmitter-relay station is to accept a smaller antenna even though some loss in performance might be sustained. If a corresponding increase in operating frequency could accompany the reduction in antenna diameter so as essentially to preserve the original gain and beam width of the antenna, then no performance would be lost. Other factors, such as search, acquisition, and tracking, would need to be considered in a given situation. For the present, the effect of substituting an antenna one-half as large on total weight in orbit was investigated without presuming any increase in frequency or power. The results are given in figure 13(b). The total weights of all unmanned communication stations with the smaller antenna are shown to be well within the launch capabilities of the Saturn IB or the Titan IIIC vehicles at either orbital altitude.

CONCLUDING REMARKS

The objectives of the present study were (a) to point out potential advantages of using an orbiting deep-space communications station primarily for continuously tracking and monitoring spacecraft engaged in interplanetary missions; (b) to assess the feasibility of the concept; and (c) to define problem areas in which further research and development would be necessary or particularly advantageous.

First, the advantages found to be worth consideration are:

(a) The ability of a single orbiting satellite to maintain continuous communications with an interplanetary spacecraft in contrast to the requirement of multiple (3 or more) ground-based stations for this purpose.

(b) Advantages resulting from the use of communications frequencies higher than those available to ground-based stations subject to atmospheric attenuation; these advantages can result in greater information rates and/or reductions in size and weight of communication equipment and of power supplies on board interplanetary spacecraft.

(c) The enabling of concurrent deep-space missions to be carried out with greater assurance of continuous tracking and monitoring by two independent communications systems, one in Earth orbit, the other the ground-based DSIF network.

(d) The capability to perform other useful functions in which uninterrupted observations are particularly advantageous; examples are observation of galactic radio sources over a relatively wide spectrum of frequencies; a search for galactic intelligence; use

as a navigation beacon for spacecraft returning to Earth from deep space; and use as a calibration device for ground-based antennas.

Secondly, no insurmountable problems, electronic, orbital, structural, logistical, or of other nature, were found to preclude the securing of the cited benefits and advantages of an orbiting radio communications station.

Finally, although it is concluded that no technological break-throughs are required to develop an operational orbiting radio station of one or another of the types described in this study, a number of problems were found to warrant further study and experimentation. Included among these are the following:

(a) Construction and packaging of lightweight, high-gain antennas for use in Earth orbit. The present study indicates that the weight of antennas as large as 26 m in diameter and of an all-metal fold-out type of construction constitutes over one-half of the orbital weight of an unmanned relay station and more than one-third of the weight of an unmanned transmitter-relay station. In addition, means must be found for reducing the minimum diameter of the folded antenna in its launch configuration to adapt it to the dimensions of certain launch vehicles (e.g., the Titan IIIC) which otherwise could in some instances place an unmanned station into orbit. The possibility of reducing weight and package volume of high-gain antennas by other methods of construction should be explored, since the potential benefits are large. Inflatable structures may be the answer.

(b) Communications optimization. A thorough study of the communications problems characteristic of an orbiting antenna designed to track and communicate with another antenna in deep space should be made to determine the most advantageous combinations of communications frequencies and sizes of both antennas within the limitations imposed by tracking capabilities, by antenna structural problems, by availability or suitability of key electronic components, and by other constraints. In the present context, "most advantageous" combinations are considered to be those which will maximize the ratio of information rate to the interplanetary spacecraft on-board weight associated with transmission of data. For other uses of the stations, such as observation of galactic radio sources, other criteria would be used to obtain the greatest advantage.

(c) Attitude control over long periods of time. The use of cold-gas jets over periods as long as 6 months for attitude control of communications satellites of the types considered here would require large weights of gas and associated tankage. This weight penalty could be reduced if small chemical rockets could be used. Such engines would have to be capable of thousands of restarts to be useful in this application. Development of engines with this capability would be of most benefit in the case of the simple relay station. In the case of the more useful transmitter-relay

communications satellite, for which a large power supply is a requirement, the use of low-thrust ion engines currently being developed could reduce propellant-weight requirements for attitude control to less than 5 percent of those based on the use of chemical engines. Problems associated with the use of ion thrustors for this purpose would no doubt require attention. Another problem deserving study is the minimization of gravity-gradient torques by proper design of the orbiting radio station.

(d) Large power supplies for extended use. The inclusion of a capability of transmission of voice communications to manned interplanetary spacecraft or of command signals to distant unmanned interplanetary probes appears desirable. In this case a power supply capable of producing a maximum output of about 150 kW is likely to be required to meet the probable extreme needs (voice communication over a distance of about 10 A.U.). Current development of nuclear power sources for space applications could probably be continued to produce a power source of the requisite level and suitable specific power for the presently conceived purpose. The availability of a large power source could made the use of ion engines practical not only for attitude control as mentioned earlier, but also for orbital maneuvers which are required between and perhaps during interplanetary missions.

(e) Orbit selection.- Although communications satellite orbits suitable for the application to an example manned Mars mission were found in the present study, no attempt was made to insure that the orbits were optimal in the sense that weight penalties for radiation shielding or for orbital maneuvers were least. Also, other interplanetary missions may require station orbits which involve more extensive orbital changes or higher altitudes than the example mission to Mars. Additional study is needed of satellite orbits and their requirements for a range of likely missions, both manned and unmanned, throughout the solar system.

National Aeronautics and Space Administration
Moffett Field, Calif., Aug. 3, 1965

APPENDIX

SYMBOL DEFINITIONS

A_T, A_R	actual areas of transmitting and receiving antennas
c	speed of light
D	diameter of the antenna, m
f	frequency
h	orbital altitude
h_{ion}	assumed altitude of disturbing ionosphere
h_{min}	minimum altitude at which occultation is avoided
I	obliquity of the ecliptic
I_c	moment of inertia about c-axis, sketch (a)
I_g	moment of inertia about g-axis, sketch (a)
i	inclination of orbit to equatorial plane
K_T, K_R	transmitting and receiving antenna design constants
K-band	10.9 - 36 GHz
l	heliocentric longitude in the ecliptic plane
l_i	initial value of l
P	perpendicular to the antenna orbit
P_T, P_R	transmitted and received power
Q-band	36 - 46 GHz
R	distance between transmitting and receiving antenna
$R_\oplus$	radius of Earth
T	time after launch of the interplanetary spacecraft from Earth orbit
W_a	weight of antenna structures
X-band	5.2 - 10.9 GHz

δ angle to the instantaneous line of sight between the center of the Earth and the target vehicle

δ_o value of δ at the intercept of the dotted line and the δ-axis on figure 3

η complement of angle of inclination of antenna orbit with respect to the ecliptic plane

θ total look angle

$\dot{\theta}$ rate of change of look angle

φ angle by which the line of sight to the vehicle leads or lags the perpendicular to the antenna orbit

Ω right ascension measured from the vernal equinox ♈ in the equatorial plane

Ω_i initial right ascension of the ascending node of the antenna orbit

$\dot{\Omega}$ rate of precession of the line of nodes in the equatorial plane

♈ first point of Aries (the vernal equinox)

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TABLE I.- COMPONENT WEIGHTS

Antenna diameter	25.9 Meters			
Mode	Receiver relay unmanned	Transmitter-Receiver Relay		
		Unmanned	Manned open cycle life support	Manned partially closed life support
Antenna (all-metal, fold-out)	3900 kg	3900 kg	3900 kg	3900 kg
Equipment module	1140	1450	1590	1590
Manned living module	---	---	1590	1590
Life support and thermal control	23	230	11,360	7260
Crew - 4 men	---	---	450	450
Attitude-tracking inertial system	140	140	290	290
Electrical power	110	2520	2730	2730
Communications	180	910	950	950
Rendezvous and docking system	110	110	140	140
Antenna launch windshield	2270	2270	2270	2270
Booster attachment	140	160	180	180
Shielding 1080 km	0	0	18,600 for 90 days	18,600 for 90 days
Shielding 1300 km	0	0	68,800 for 90 days	68,800 for 90 days
Propellants for chemical-thrust 1080 km	1060	1750	7860	7090
Propellants for electrical thrust 1080 km		110	500	450
Propellants for chemical thrust 1300 km	215	355	3530	3370
Propellants for electrical thrust 1300 km		17	165	158

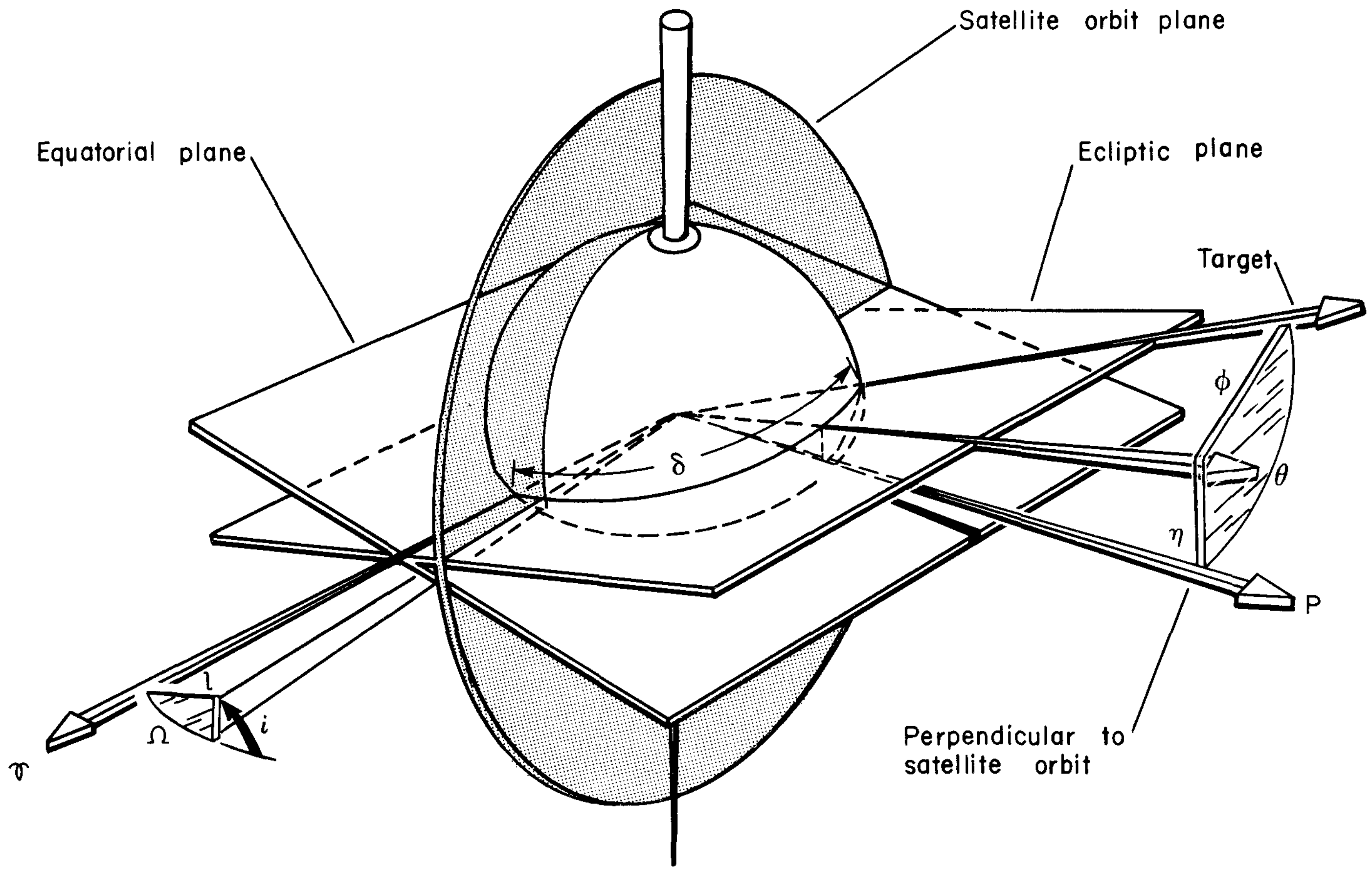
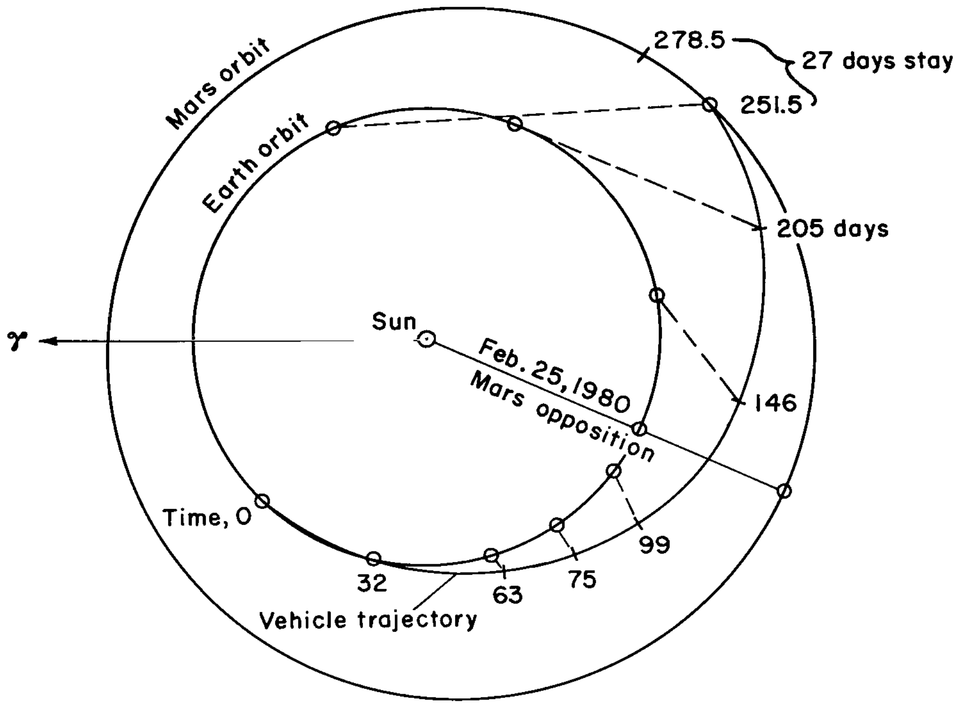
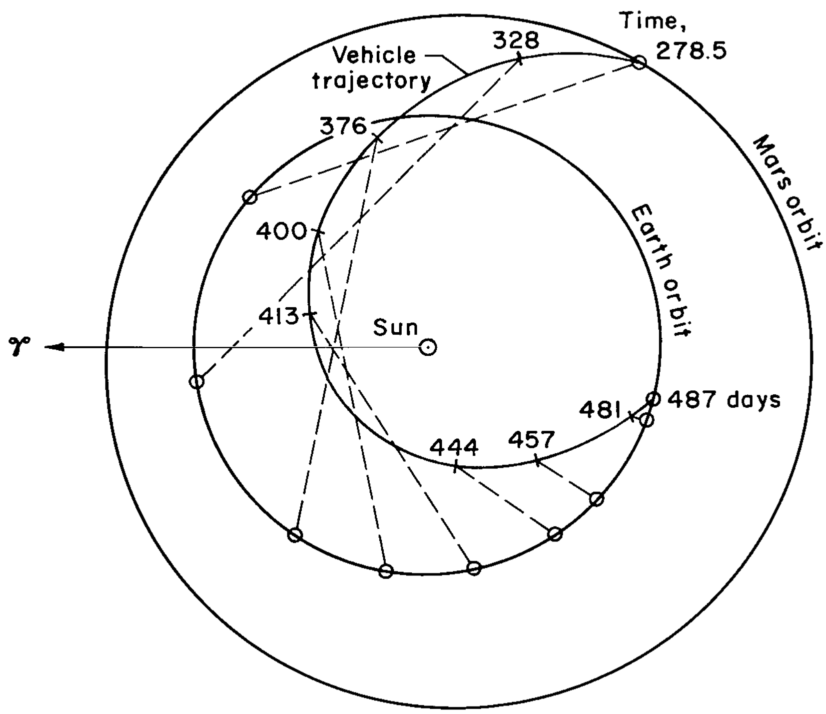


Figure 1.- Geometry for orbit selection.



(a) Earth to Mars.



(b) Mars to Earth.

Figure 2.- Typical trajectories for 1980 manned Mars mission with 27-day staytime.

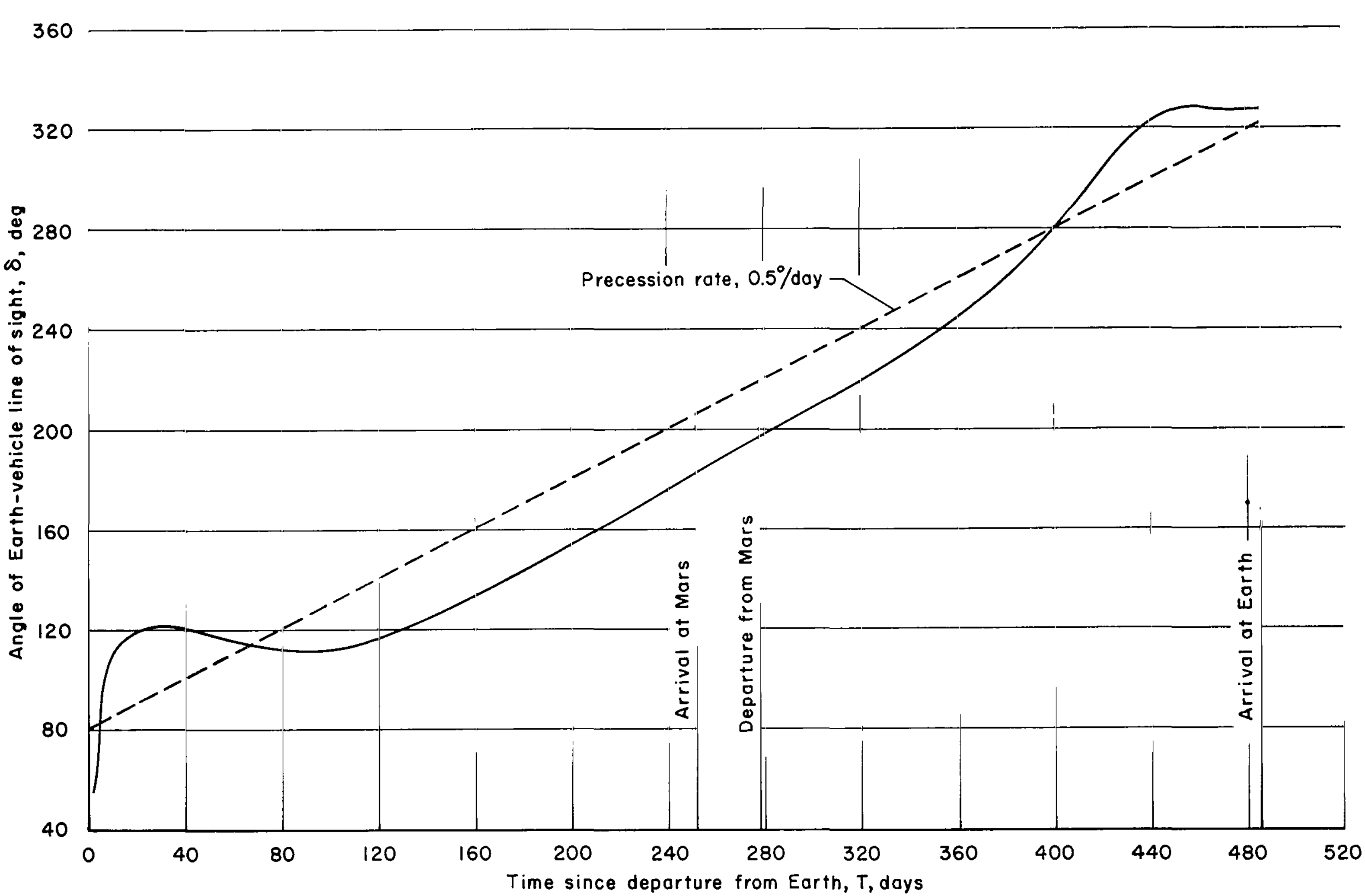


Figure 3.- Variation of Earth-vehicle line-of-sight angle with trip time.

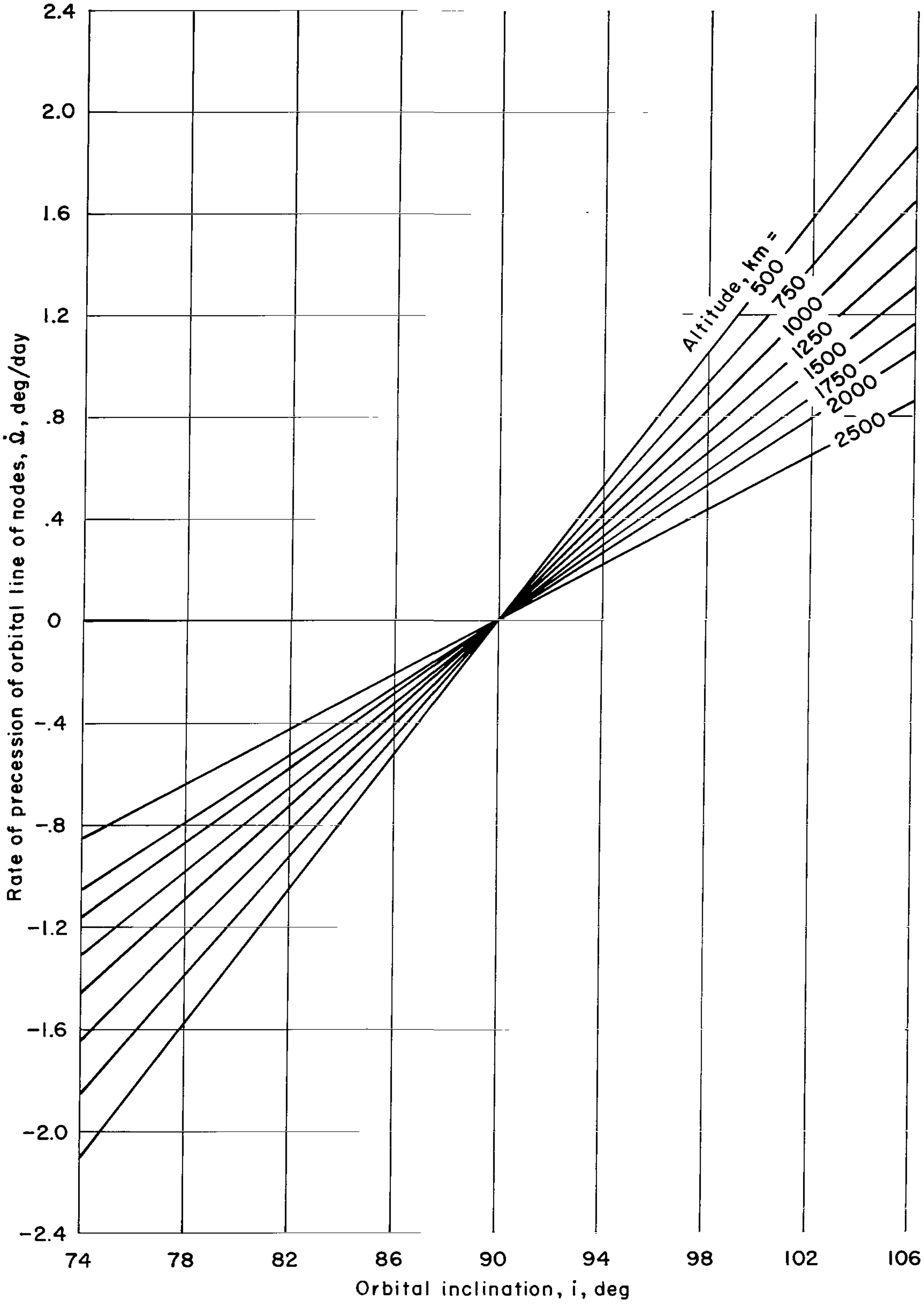


Figure 4.- Rate of precession for circular satellite orbits.

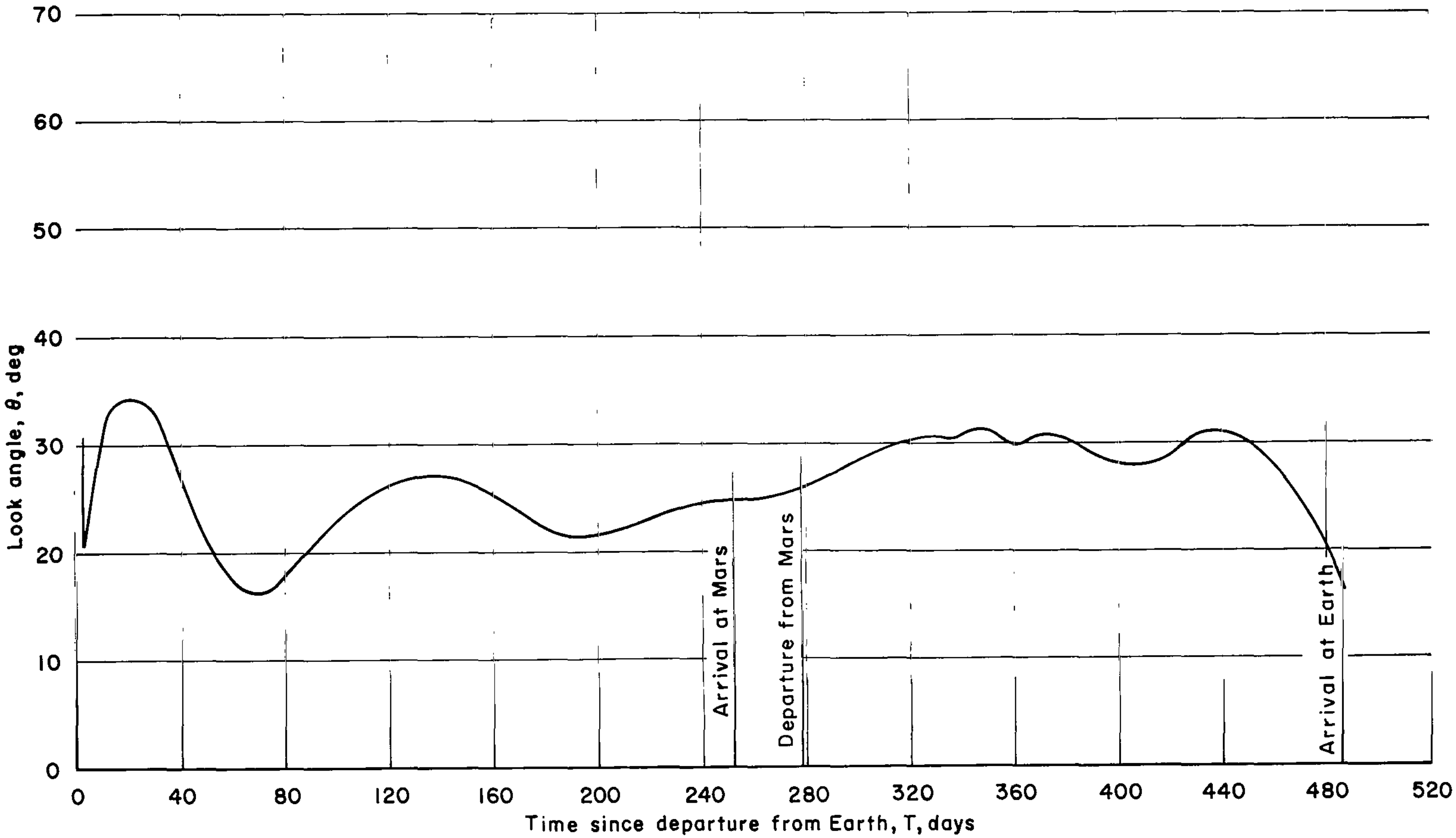


Figure 5.- History of look angle, θ, during example mission.

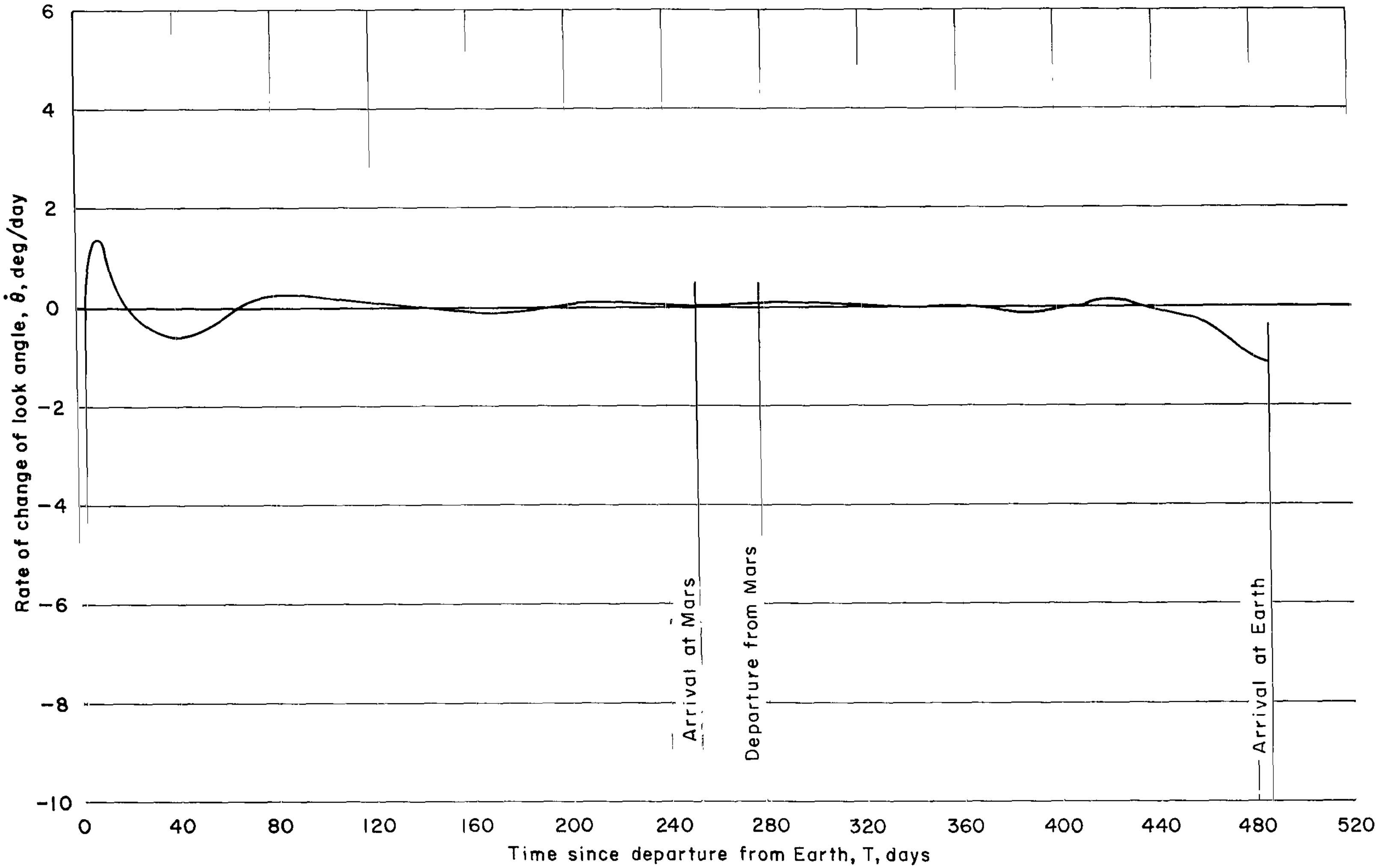
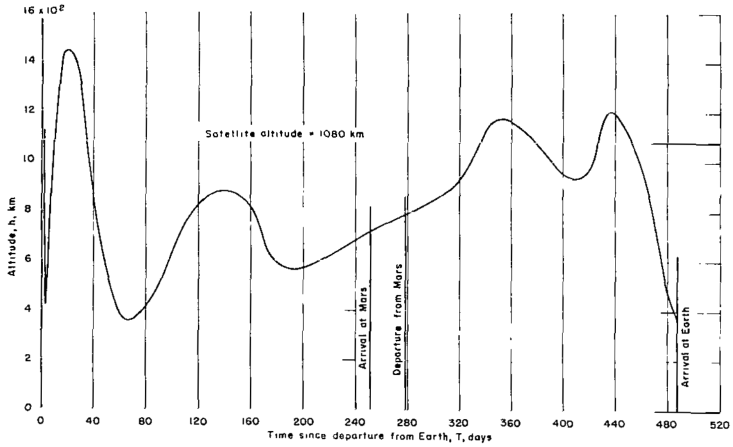
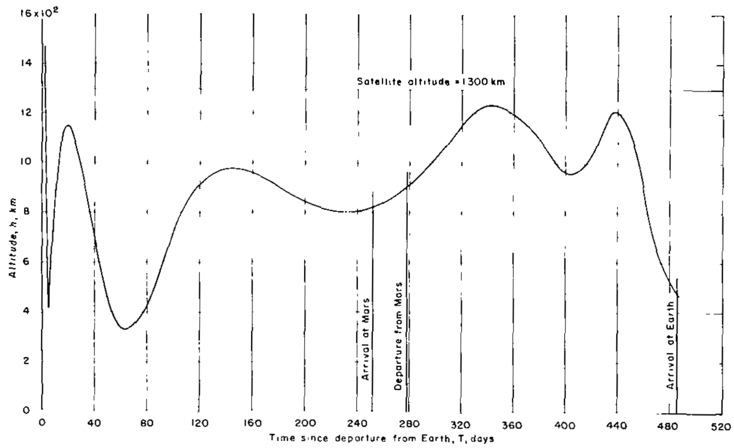


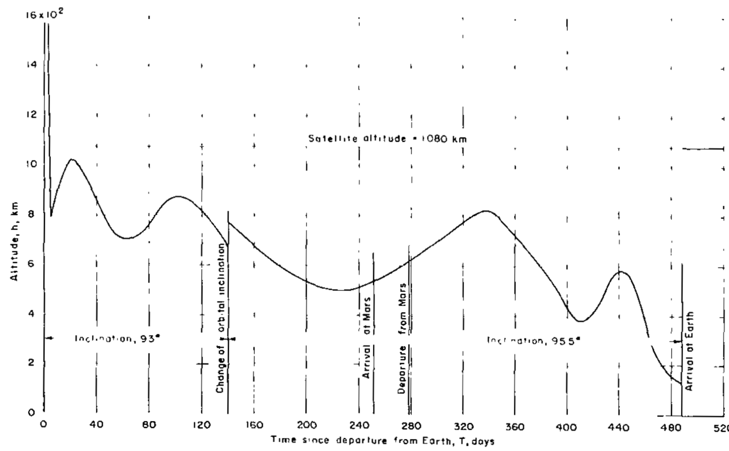
Figure 6.- History of rate of change of look angle, $\dot{\theta}$, during example mission.



(a) Orbit altitude = 1080 km; orbit inclination = 95°.



(b) Orbit altitude = 1300 km; orbit inclination = 95.4°.



(c) Orbit altitude = 1080 km; orbit inclination = initial 93°, final 95.5°.

Figure 7.- Minimum altitude to avoid occultation by Earth and its ionosphere during example mission.

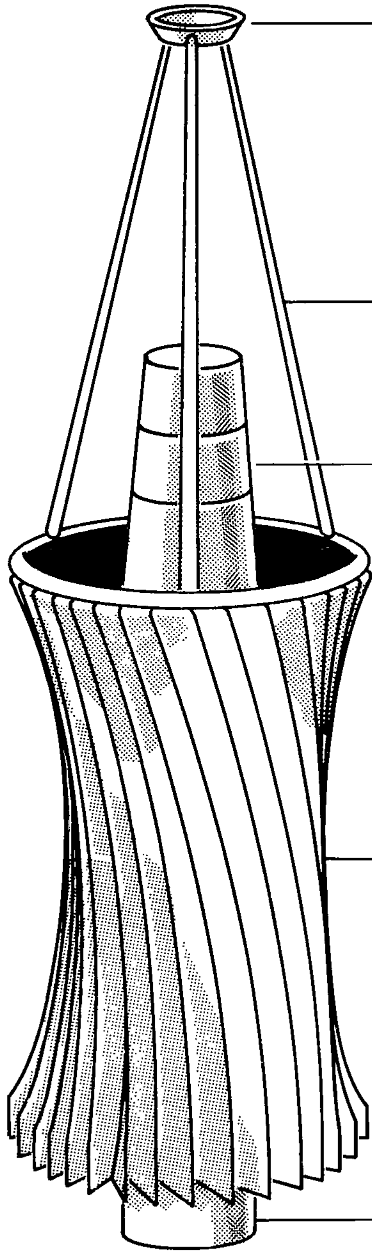


Figure 8.- All-metal fold-out antenna (folded).

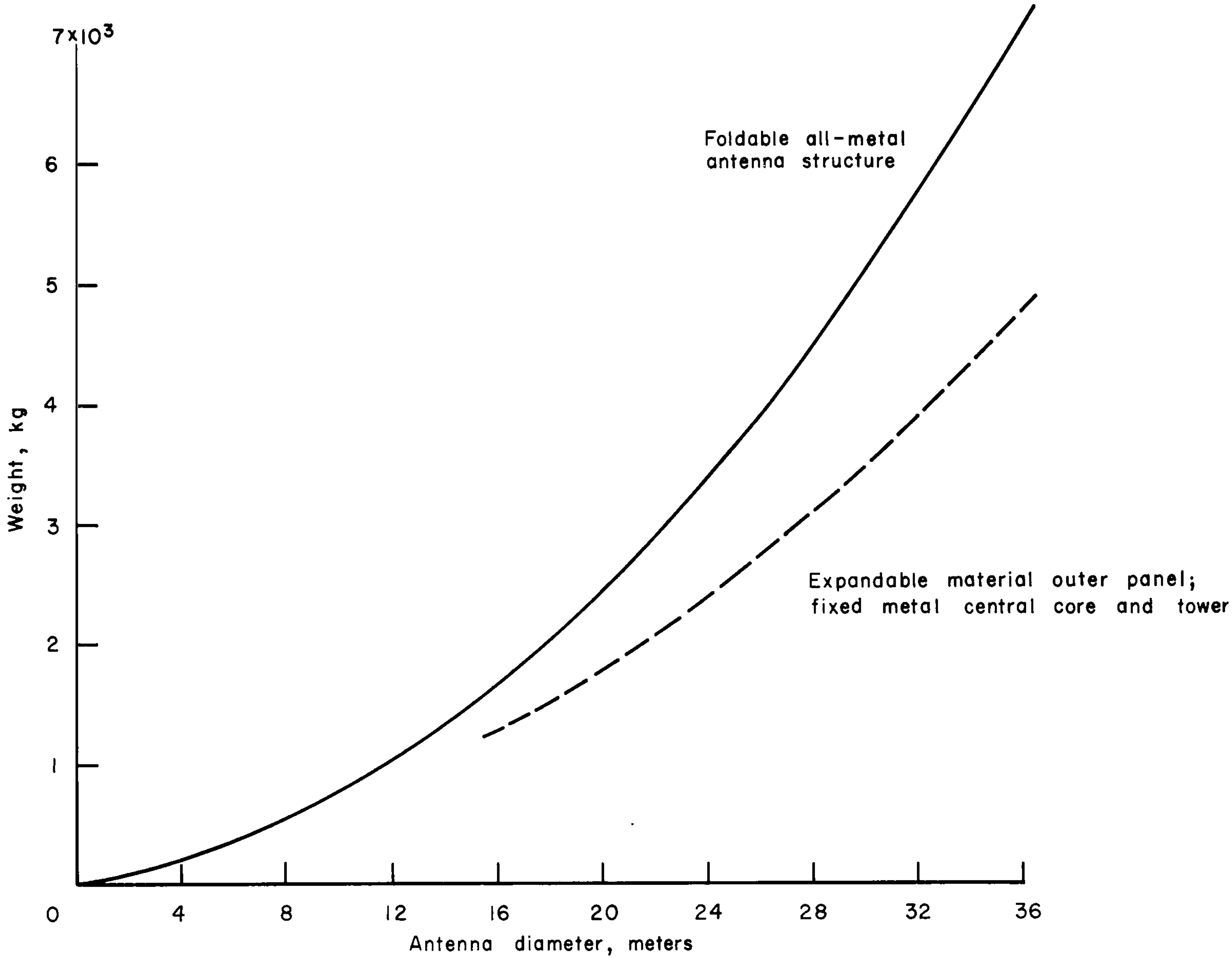
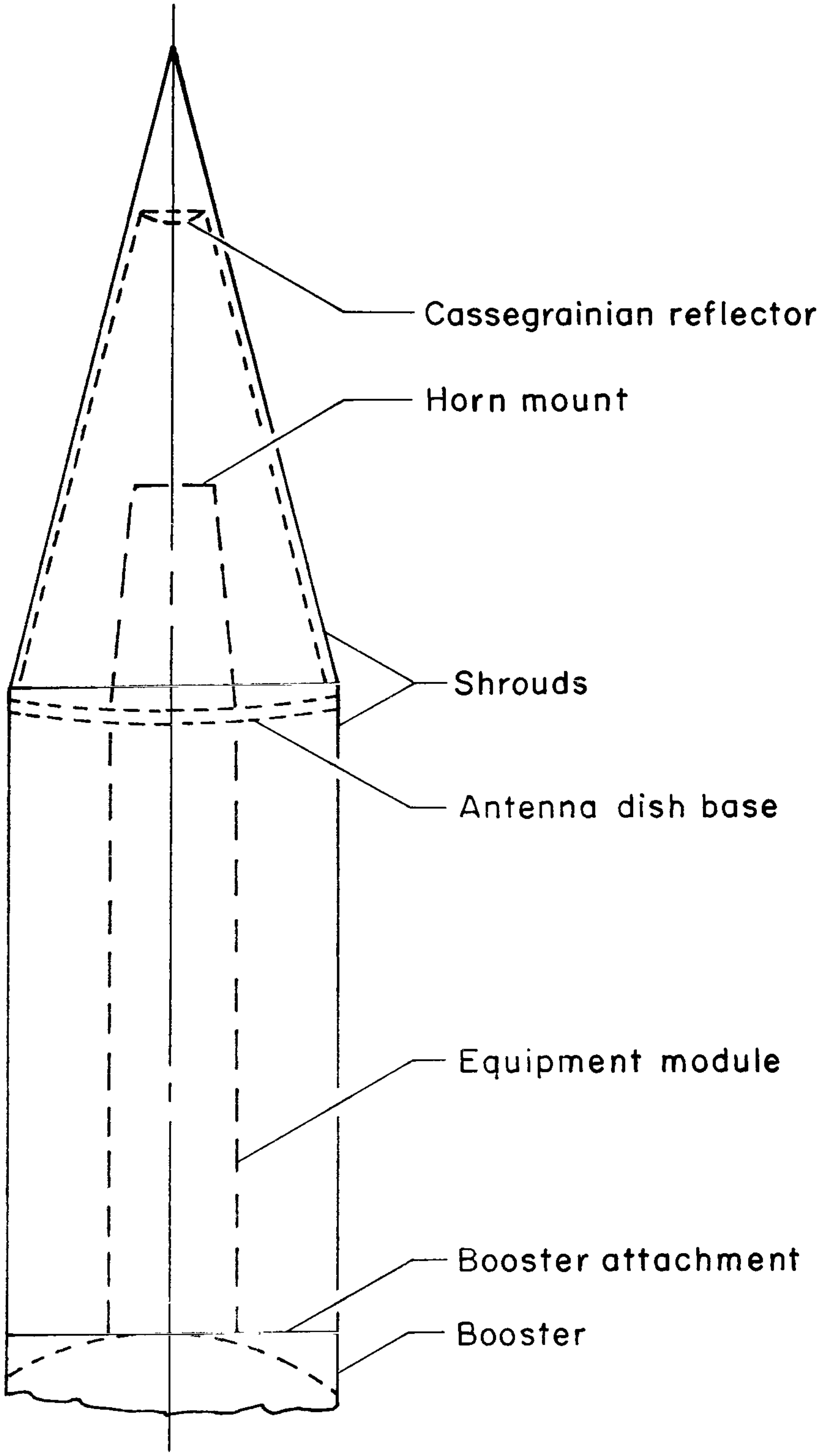
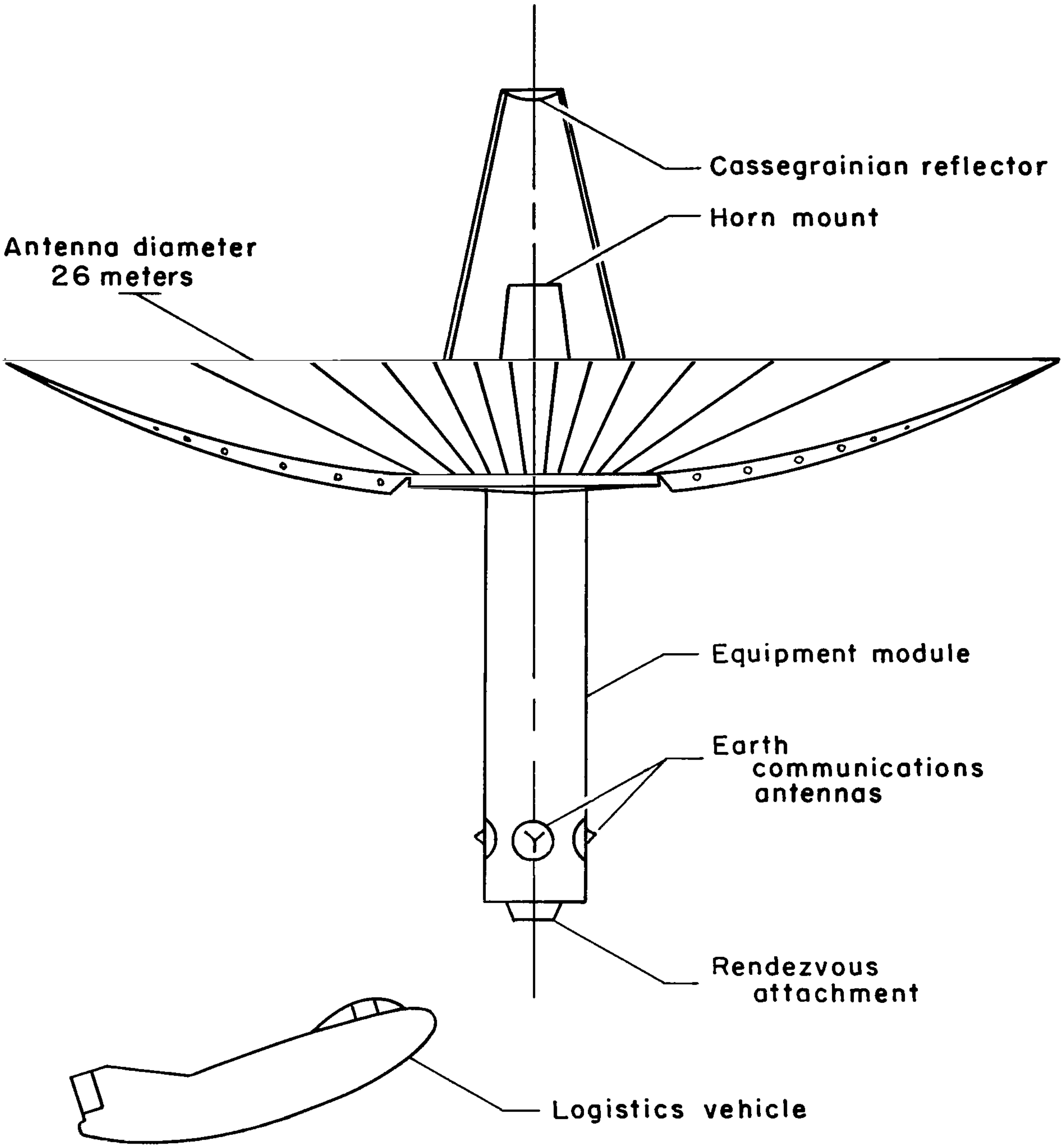


Figure 9.- Antenna weights.



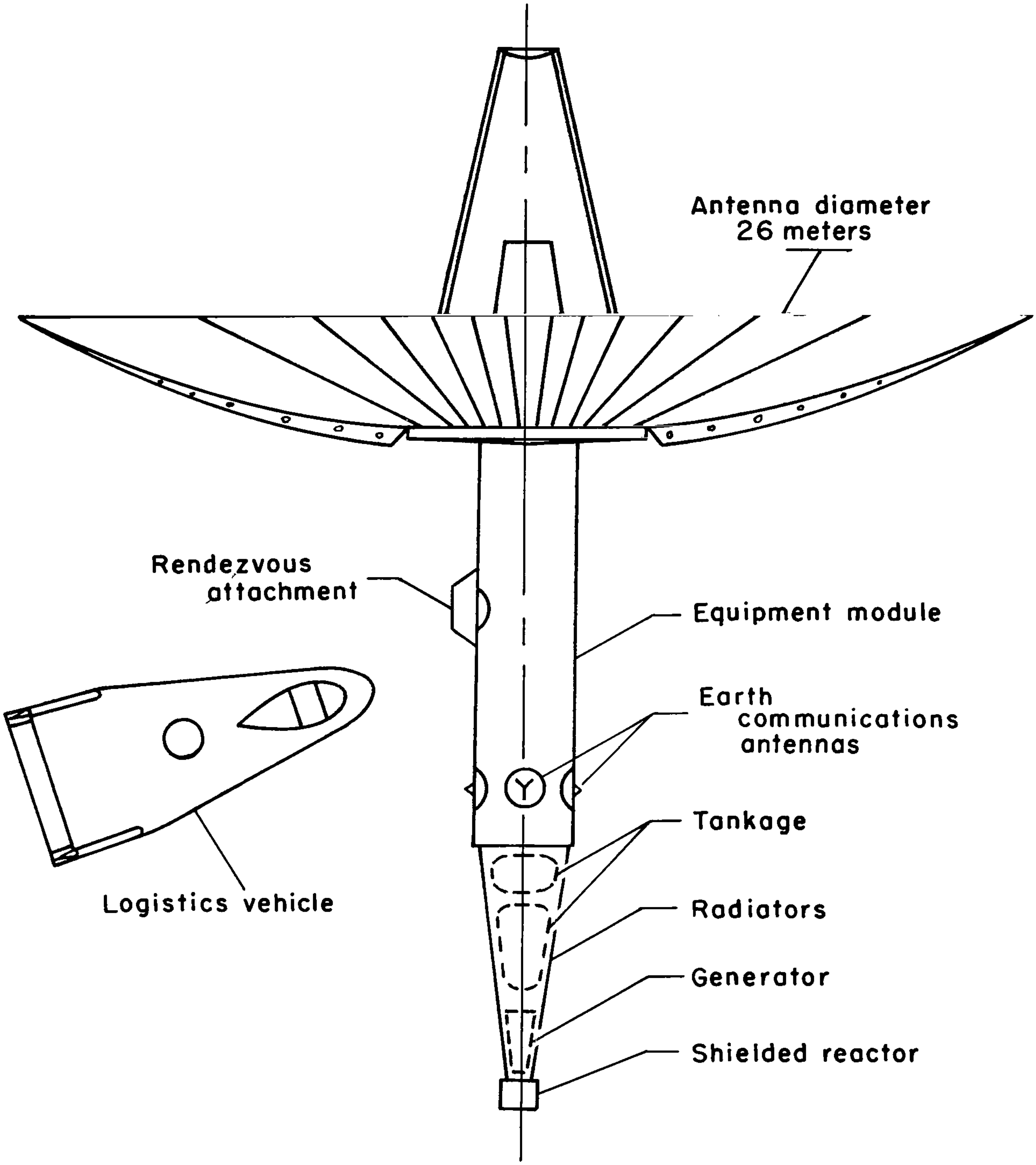
(a) Launch arrangement.

Figure 10.- Communications vehicles.



(b) Receiver-relay configuration, operational.

Figure 10.- Continued.



(c) Transmitter and receiver-relay configuration, operational.

Figure 10.- Concluded.

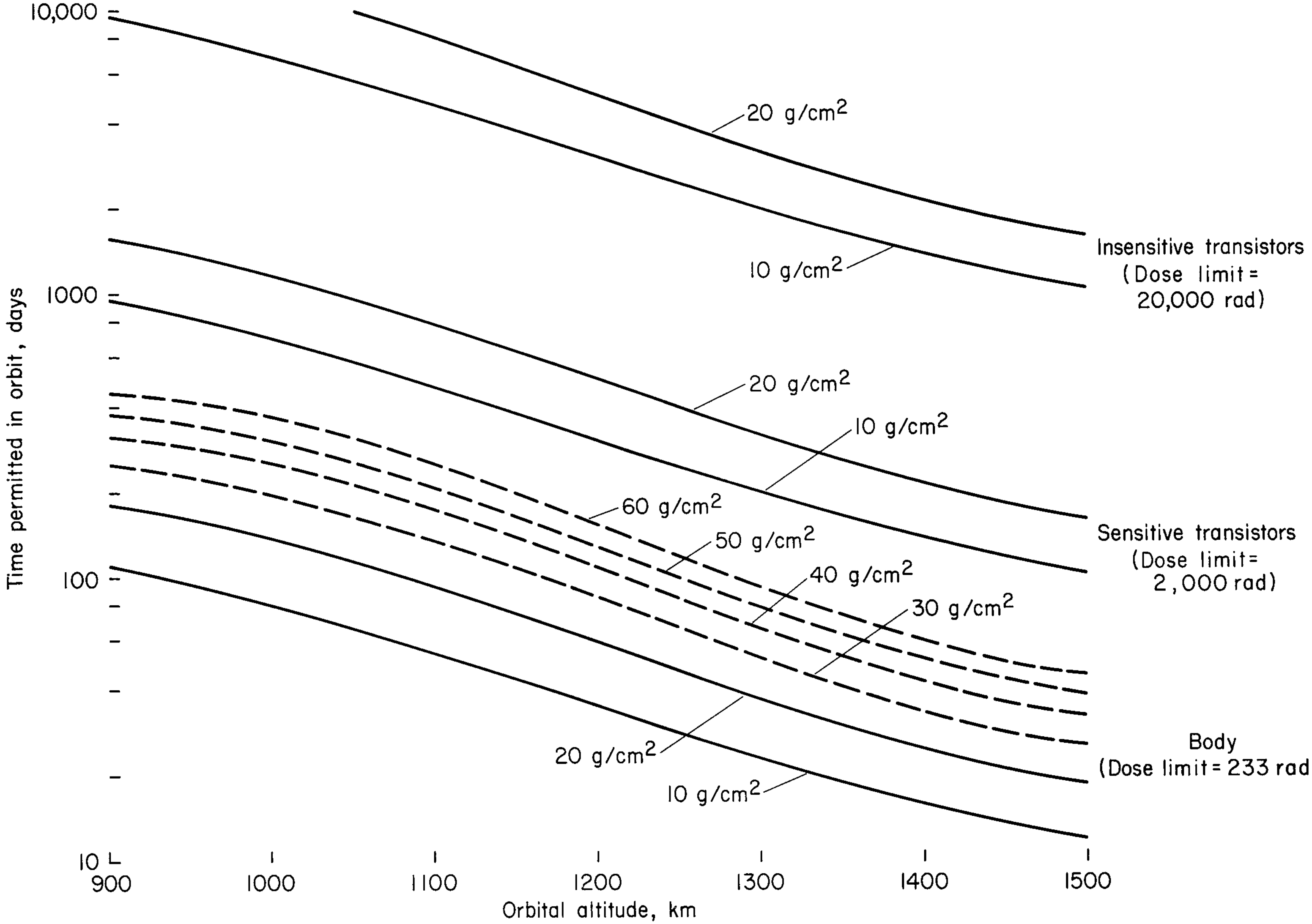


Figure 11.- Effects of altitude, shield thickness, and dosage limits on time allowed in circular polar orbits.

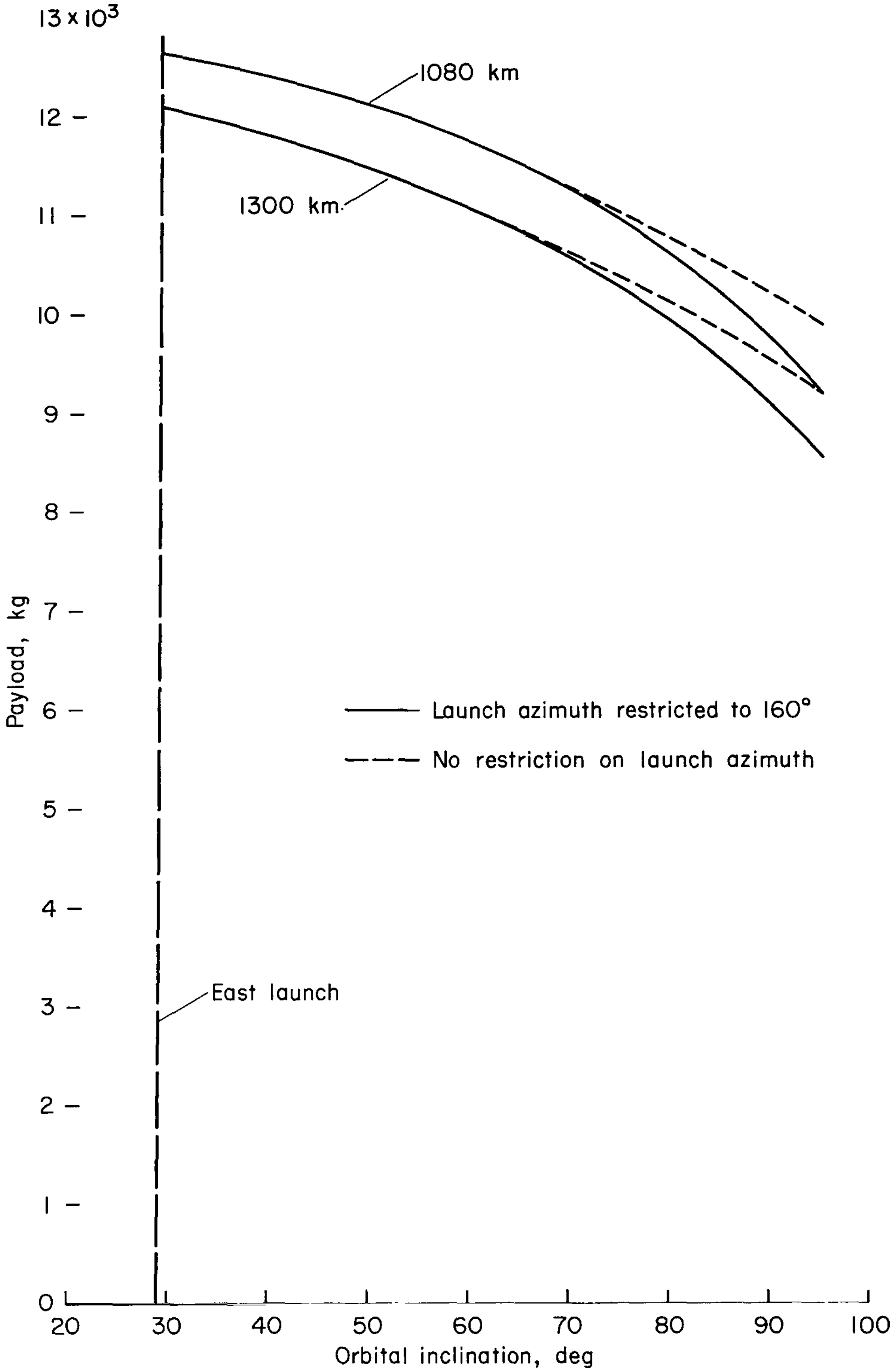
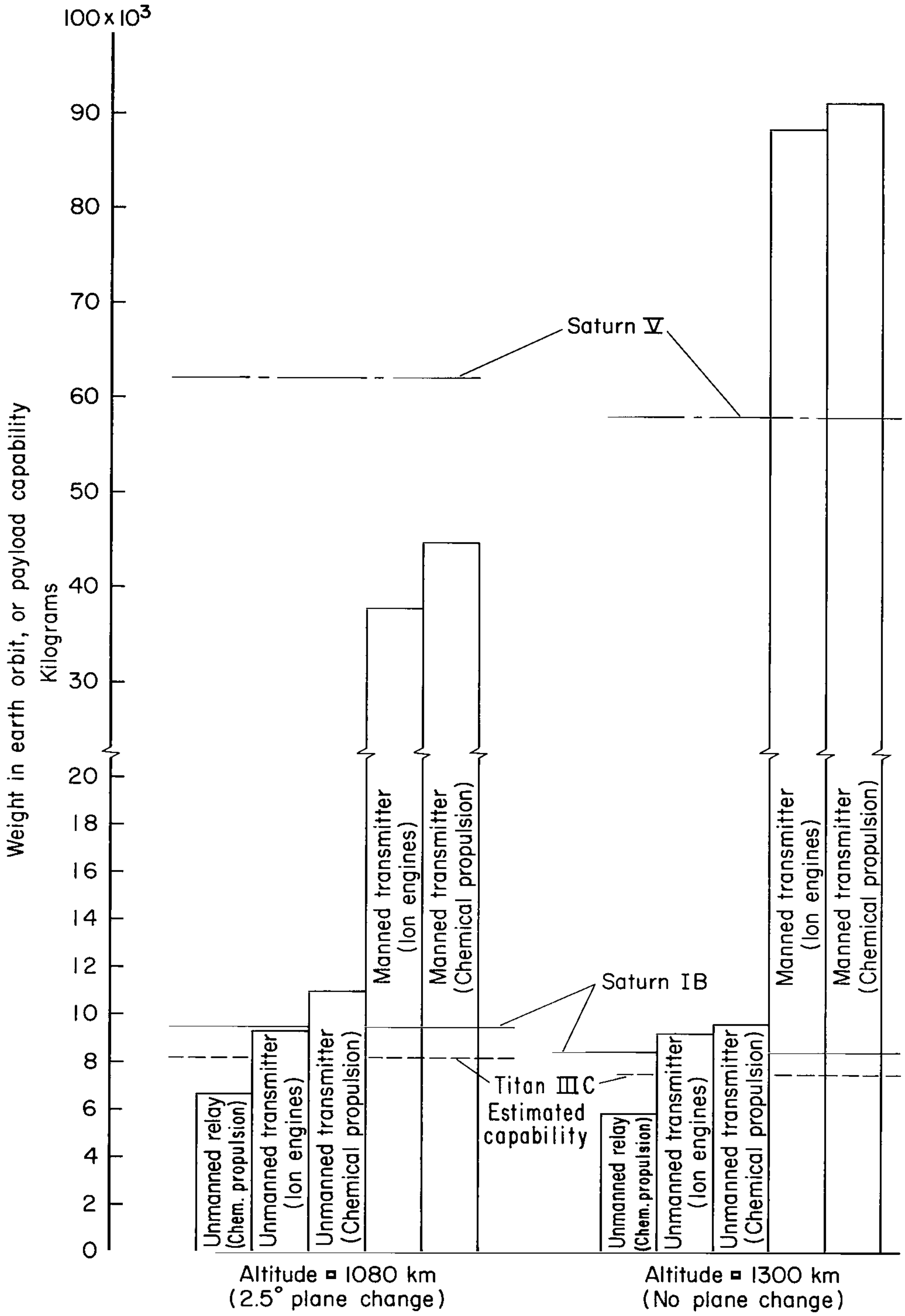
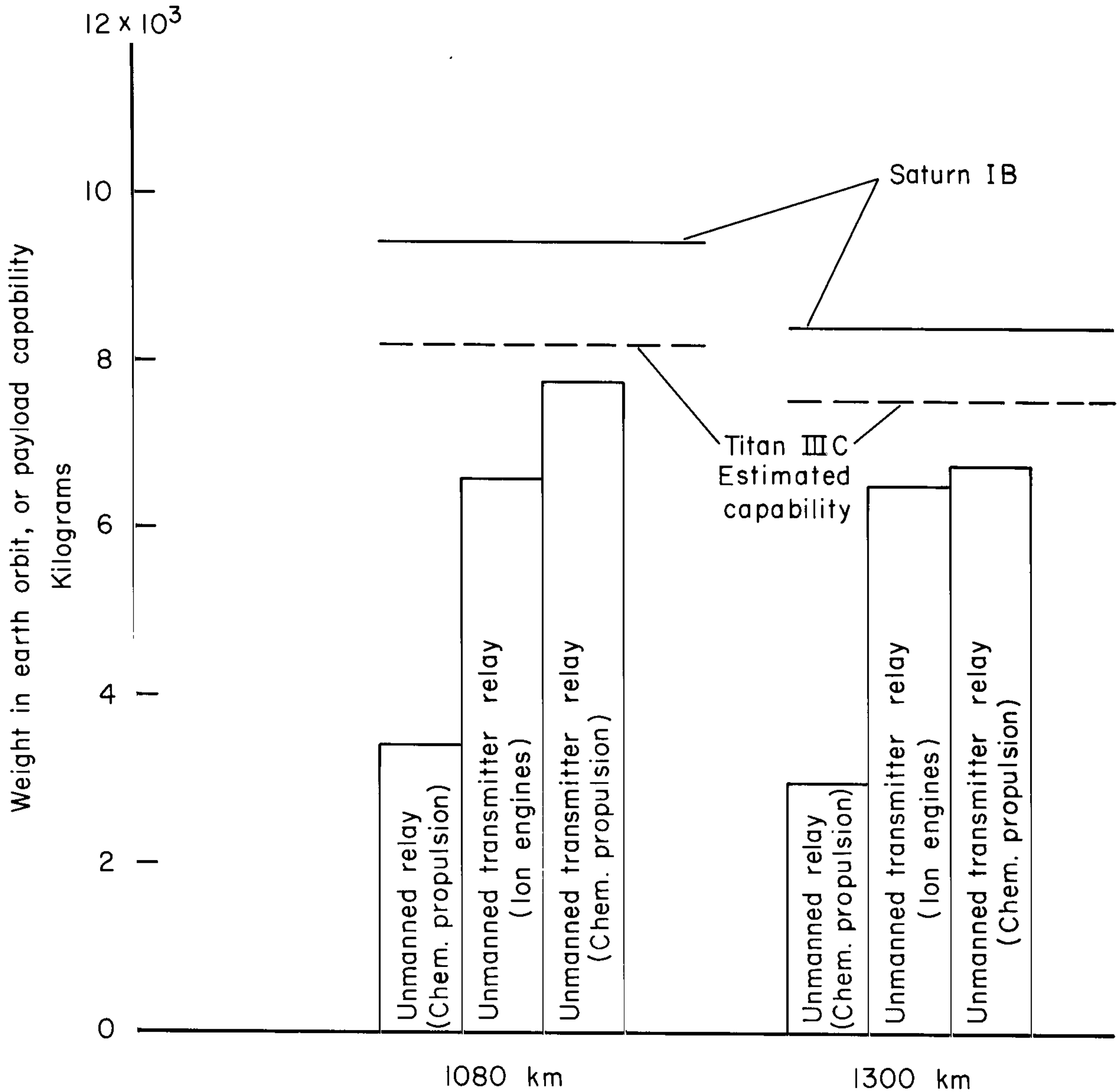


Figure 12.- Effects of orbital inclination, orbital altitude, and AMR launch restrictions upon payload capability of Saturn IB.



(a) Antenna diameter = 26 meters.

Figure 13.- Comparison of orbital weights and launch-vehicle payload capabilities.



(b) Antenna diameter = 13 meters.

Figure 13.- Concluded.

NASA-Langley, 1965

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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